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13. ABSTRACT (Maximum 200 words)

This report describes work carried out at the Gas Turbine Laboratory at MIT during the period 10/20/92 – 10/31/95, as part of a multi-investigator effort on unsteady and three-dimensional flow phenomena in turbomachines. Within the overall project, three separate tasks are specified. These are, in brief: I. Characterization of Fully Scaled, Unsteady Film Cooled Turbine Blade Heat Transfer and Aerodynamics; II. Flow Control in Compression Systems by Viscous Flow Removal; and III. Unsteady Phenomena in Turbomachinery Endwall Flows.

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PERIOD OF
INVESTIGATION: October 20, 1992 - October 31, 1995

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1. SUMMARY

1.1 Introduction and Objectives

This document constitutes the final report on a multi-investigator research program on unsteady and three-dimensional flow phenomena in turbomachines. The unifying theme in the work described is the view that primary barriers to achieving increased overall performance of turbomachines are linked to local phenomena which are inherently three-dimensional and/or unsteady. The objectives of this multi-investigator program are as follows:

1. Elucidate the structure of detailed physical phenomena (vortices, shock waves, etc.) in cooled high pressure turbines and quantify their importance to heat transfer and aerodynamic performance.

2. Examine, experimentally and computationally, new methods for suction enhancement of the pressure ratio of high performance compressor stages and effects of suction on compressor efficiency, and quantify the benefits to engine systems of the use of fluid extraction in the compressor.

3. Develop quantitative (i.e., predictive) understanding of the key issues that relate to unsteady flow effects in multi-stage turbomachines, including strategies for managing these effects, clarification of the physical phenomena involved, and quantification of their impact on performance.

1.2 Research Abstracts

The influence of inlet temperature distortion typical of combustor exits flows on turbine heat transfer has been investigated experimentally and computationally, using a highly loaded transonic turbine. Computational investigations clearly revealed relevant flow phenomena: preferential migration of hot/cold fluid leading to higher time average local surface temperature non-uniformities, radial displacement of thermally stratified fluid in the centrifugal field of the rotor, and "wobbling" of the nozzle guide vane hot streak due to unsteady blade row interaction.

The effect of suction through the blade tip in a transonic compressor has been studied experimentally and computationally . The computations showed a small increase in efficiency, but an increase in clearance flow. A fan stage has been designed using boundary layer suction, with design pressure ratio considerably above that which can be achieved without suction. Shock-boundary layer interaction control by suction is currently being investigated.

A new analysis has been developed to assess the influence of non-uniform tip clearances on compressor performance and stability. In particular, a rigorous hydrodynamic stability analysis has been carried out of the non-uniform unsteady flow field that results from an asymmetric clearance, such as would be generated by compressor case deformation. Experiments are being carried out at an engine company facility to assess the predictions of the theory. The effect of the downstream stator pressure field on rotor tip clearance vortical structure is also being examined.

2.0 NEW FINDINGS AND ACCOMPLISHMENTS

The introduction of 8% momentum turbulence at stage inlet was found to have no influence on turbine rotor heat transfer. A novel mechanism associated with turbine rotor-stator interaction, which leads to time averaged temperature non-uniformities in a turbine stage, has been identified.

First-of-a-kind computations have been carried out of the effect of (1) rotating (propagating) inlet distortions, and (2) asymmetric rotor tip clearance, on unsteady flow disturbance structure and loss in stability for a multistage compressor. The former has been assessed experimentally on a four-stage, low speed compressor. The analysis developed for the latter gives the ability to address the effect of structural deformation on aerodynamic performance, both steady-state and nonsteady.

The use of suction on compressor stator blades has been shown to have the potential to lead to increased loading capability compared to current designs, as well as to increase the overall cycle efficiency.

3. TASK I: CHARACTERIZATION OF FULLY SCALED, UNSTEADY FILM COOLED TURBINE BLADE HEAT TRANSFER AND AERODYNAMICS
(A.H. Epstein)

3.1 Introduction

Convective heat transfer is important in many aerospace applications. One of the most important is in high performance aircraft gas turbine engines. In these devices, turbine inlet gas temperature is a principal determinant of the engine's thrust to weight ratio. Since the gas temperature (currently in the neighborhood of 2000°K) is about 800°K hotter than the allowable blade temperature, heat transfer and cooling is a critical technology area. In particular, modern blade designs are dependent on film cooling in which external heat transfer to the blade is reduced by blowing an air film layer over the surface. The accuracy to which the blades can be maintained at the desired temperature distribution has a strong influence on blade life and therefore engine reliability and maintainability. An error as little as 20°C (out of 2000!) can halve the blade life. Thus, successful turbine heat transfer design requires enormous precision in order to balance efficiency (overcooling) against premature failure. Progress in this area has been impressive over the last fifty years but successful designs still depend heavily on empiricism. This adds considerable uncertainty to new designs, increases costs (to fix problems), and inhibits innovation outside the empirical data base. Current new designs often include 100-200°K in "padding" (reduced gas temperature) to protect against turbine blade over temperature problems.

The heart of the problem is the detailed fluid mechanics of the turbine (here we consider the heat transfer to be nothing more than a subset of the fluid flow). A traditional approach to making a basic contribution here is to study various aspects of the air film and boundary layer behavior in some detail. Turbine geometries and flowfields are very complex, however (Fig. 1) -- with high levels of periodic unsteadiness, moving shock waves, impinging wakes, large centrifugal forces, and very high turbulence levels about highly loaded airfoils. These large scale (relative to the boundary layer) flow features have a first order influence on the blade heat transfer and raise many questions as to how steady boundary layer studies can be applied in detail to turbine rotor blades or, indeed, if such concepts are even germane. The practical problem is that the geometric complexity and harsh environment make it extremely difficult to measure the flow in an engine turbine in detail. This same complex 3-D, unsteady nature of the flowfield is also extremely taxing to the state of the art of computational fluid dynamics (CFD).
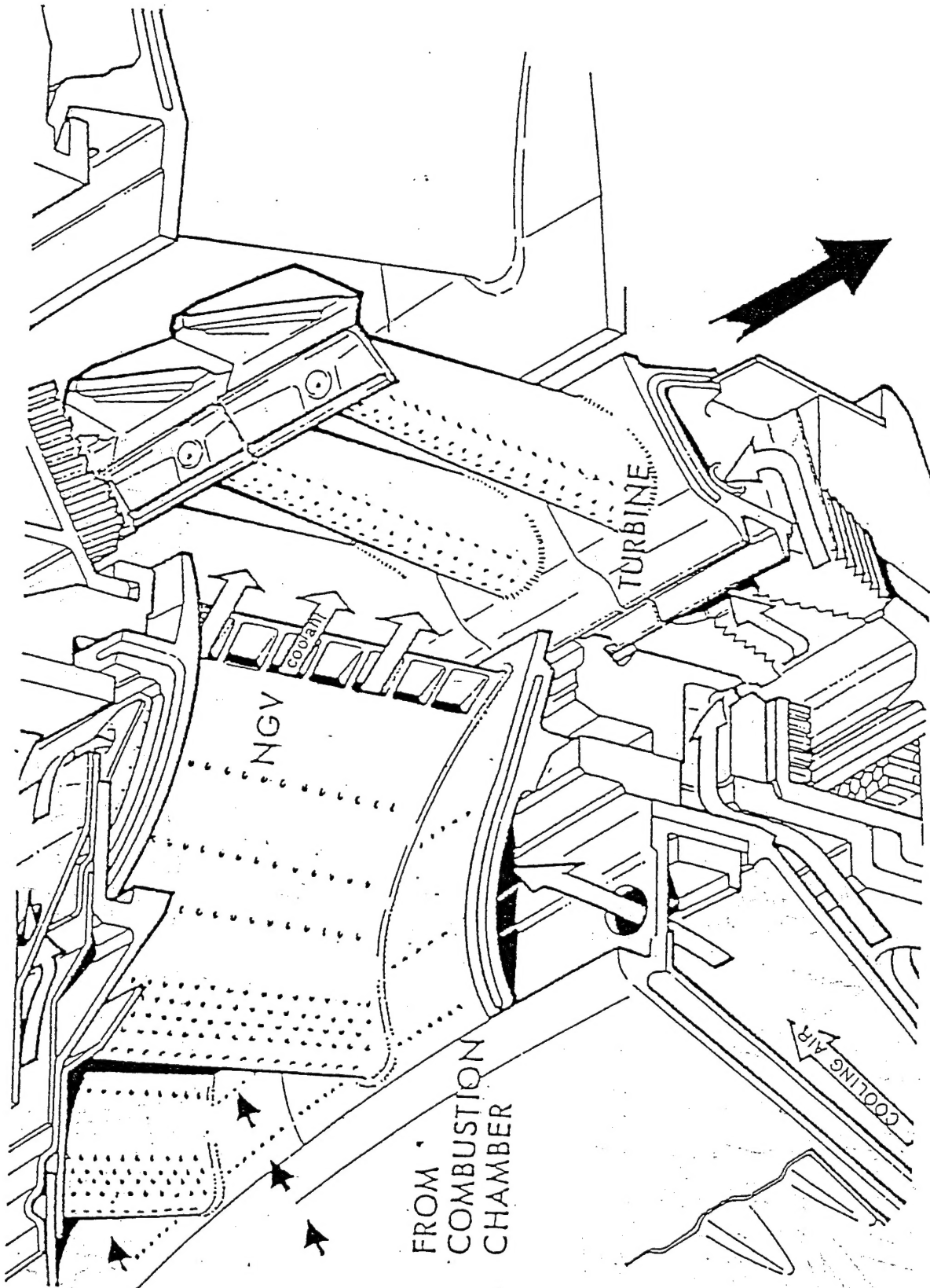
Fig. 1: Film-cooled high pressure turbine geometry.

3.2 Background on Unsteadiness in Turbine Heat Transfer

There are two principal concerns in the aerothermal design of modern aircraft engine turbines -- maximizing aerodynamic efficiency and optimizing heat transfer so as to achieve the desired turbine component life. Enormous progress has been made in both of these areas in the last 50 years and we now have highly efficient machines with what a few years ago would have been considered extraordinary life. Much of this progress has been due to improvements in analysis and computational techniques, as well as the empirical experience gained over the past five decades. It is clear, however, that the current level of understanding is far from complete and this has significant impact on turbine performance, life, and engine development and maintenance costs.

Of particular concern is the general lack of agreement between *ab initio* heat transfer predictive techniques and engine experience. This is illustrated in Fig. 2, which shows the ratio between predicted and measured heat transfer along turbine blade pressure surfaces. This information, from several companies, shows a consistent discrepancy with the implication, perhaps, that there is a fundamental flaw in our concept or modelling of the processes involved. (This figure should not be construed as implying that a company's design system, employing empirical data bases, will not give a more accurate prediction).
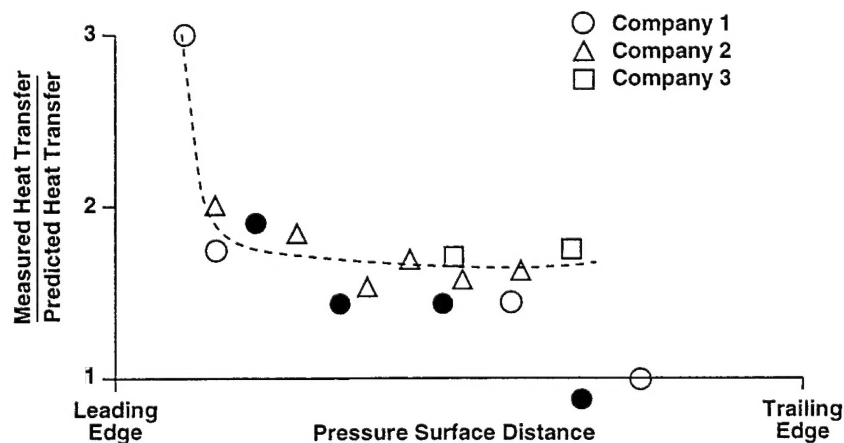
Fig. 2: Current methods underpredict engine heat transfer.

Whence does this discrepancy arise? We can start to address this question by examining the somewhat simpler flowfields about uncooled turbine airfoils. Current *ab initio* techniques can do a good job on steady two-dimensional cascade flows. Figure 3 compares the prediction of a 2-D thin shear layer Navier-Stokes code with cascade measurements and shows good agreement (Abhari *et al.*, 1992). Yet, extension of this analysis to 3-D does not agree well with engine experience, as evident in Fig. 2.
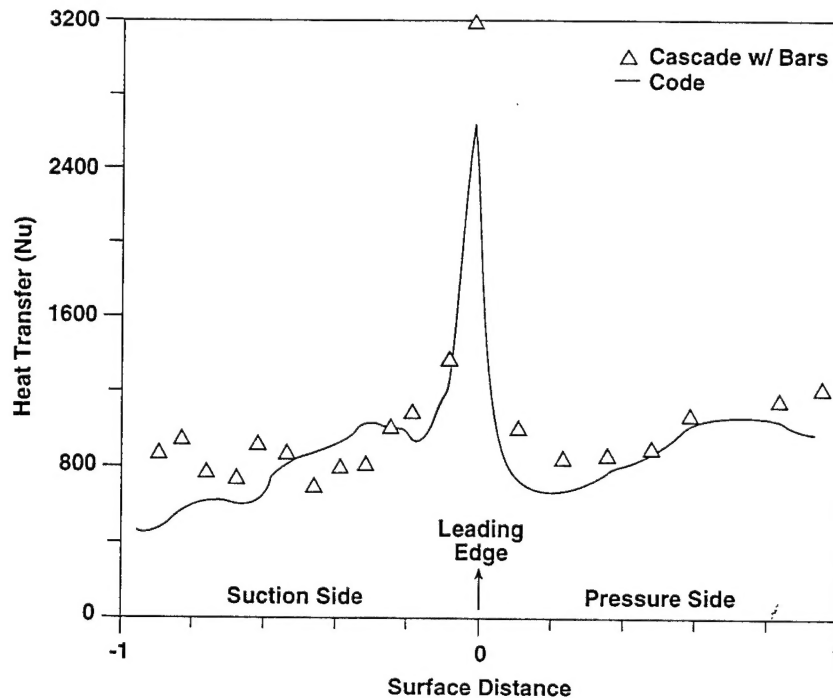
Fig. 3: Comparison of a 2-D, thin shear layer, N.S. code with turbine cascade heat transfer measurements.

Since such cascade measurements can be well predicted, it makes sense to ask "How does an engine turbine differ from a cascade?" In terms of the basic fluid mechanics, we can identify three major differences: 1) the engine has a 3-D rather than 2-D geometry; 2) the turbine in the engine rotates so there are unsteady blade row interactions, frame changes, and g field effects; and 3) the engine turbine operates behind a combustor so that the turbine inflow is spatially nonuniform and has a high level of both vortical and thermal turbulence.

The influence of these phenomena have been explored in a short duration turbine test facility at MIT and elsewhere. Here, the 3-D, rotational, and blade row interaction effects should be the same as for an engine, since both the geometry and the non-dimensional parameters (Reynolds No., Mach No.'s, Rossby No., Prandtl No., temperature ratios) which govern fluid flow are the same as in an engine. Comparison of these 3-D, rotating rig, measurements with that for the same uncooled geometry in a cascade and from a 2-D calculation show good agreement (Fig. 4). Thus we can infer that, for this turbine, three-dimensional and blade row interaction effects do not in themselves generate the principal discrepancy between cascade and engine behavior (Guenette *et al.*, 1989).

More recent studies on the same turbine with increased levels of inlet turbulence, and large scale inlet temperature nonuniformity (characteristic of combustor outflows) more closely agree
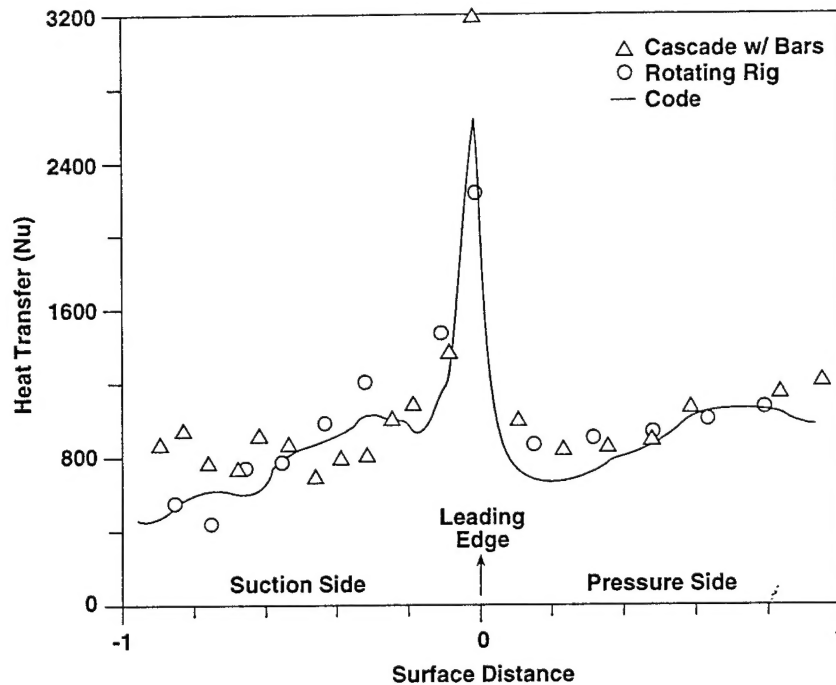
Fig. 4: Comparison of a 2-D N.S. code, cascade, and rotating rig rotor blade heat transfer measurements.

with engine rather than cascade experience (Fig. 5) (Shang, 1995). Specifically, circumferentially nonuniform hot spots at the inlet (hot streaks) increased the heat transfer preferentially to the pressure surface of the turbine rotor blades (Fig. 6). Analysis showed that these pressure surface hot spots were generated by three separate physical phenomena unique to rotating turbine stages - rotation-induced buoyancy, relative frame convection (the so-called Kerrebrock-Mikolajczak effect (Kerrebrock and Mikolajczak, 1970)), and nonuniformity in the time-averaged rotor relative inflow (Fig. 7). The first two effects are steady flow features induced by the rotor rotation and frame change. The last is an unsteady phenomenon generated by the rotor-stator interaction (specifically, the influence of the rotor passing behind the upstream vane which wobbles the vane outflow angle with a resultant nonlinear time average, as seen by the rotor). The sum of these rotational and unsteady effects is to increase the localized heat transfer by 10~40% over the area average mean value on the uncooled turbine studied.
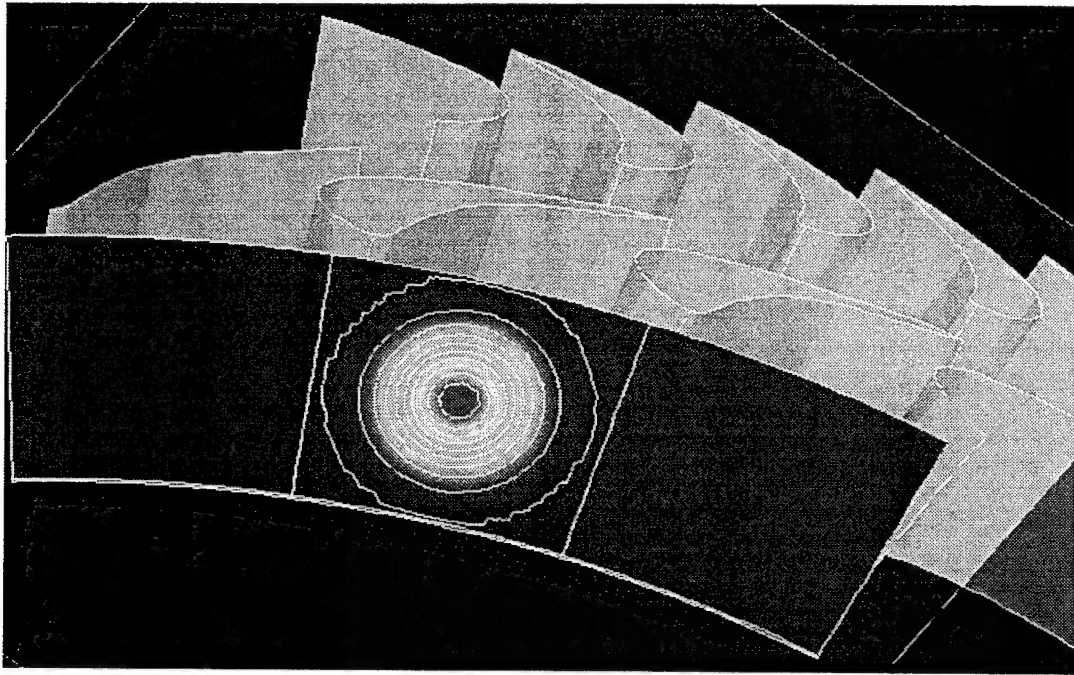
Fig. 5: View of transonic turbine stage geometry showing hot spot total temperature contours at the vane inlet plane as studied experimentally and computationally.
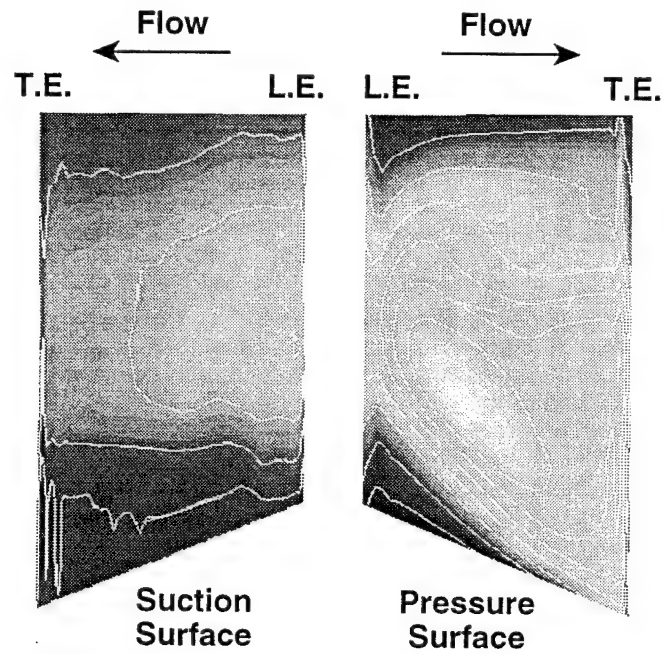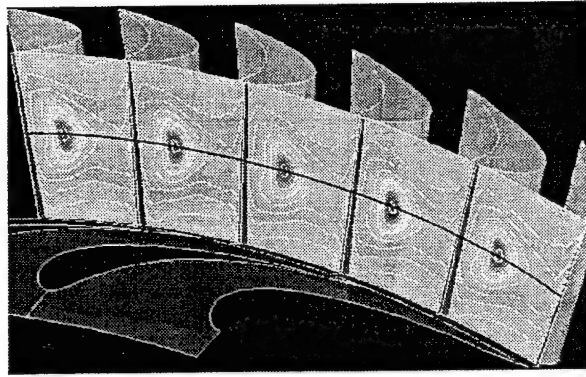

Fig. 6: Calculations show that hot streaks generate localized rotor blade hot spots on the rotor blade pressure surface (contour intervals of 10% gas-wall temperature difference).

Rotor relative time-averaged total temperature
contours at rotor leading edge plane.

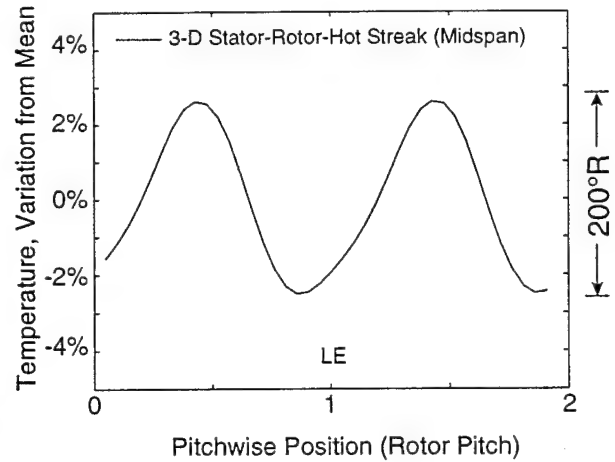Fig. 7: The time-averaged rotor relative total temperature is circumferentially nonuniform due to unsteady interactions between the rotor blades and the upstream stationary vane row. This nonuniformity can increase the local heat transfer on the blade pressure surface by 10~30%.
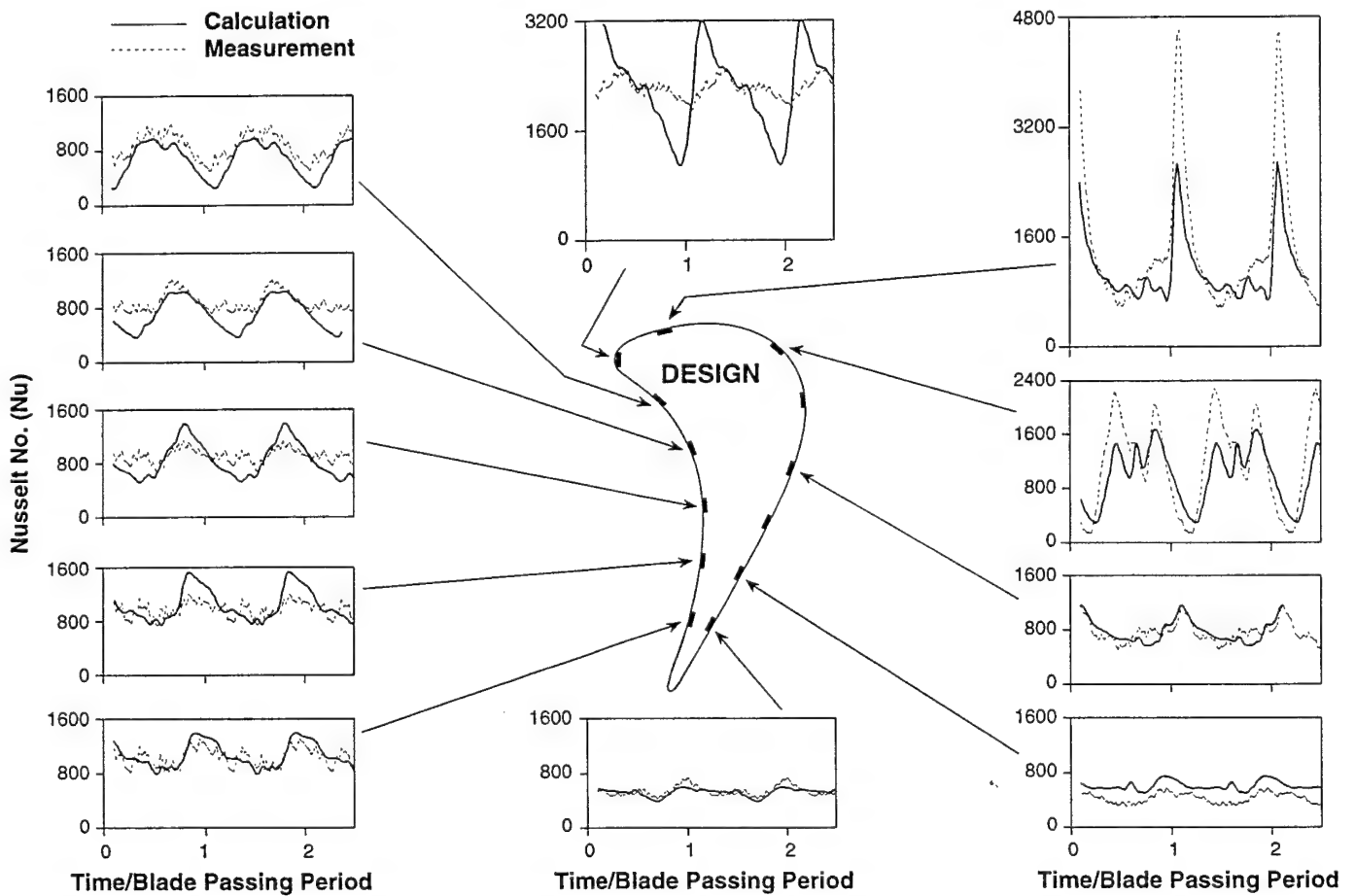

Fig. 8: Time-resolved heat transfer measured about the ACE rotor blade in the MIT blowdown turbine facility. Note that the flow everywhere is unsteady.

3.2.1 *Comparing Turbine Data and Calculation*

The agreement between steady state calculations and time-averaged data shown in Fig. 4, while impressive for an *ab initio* heat transfer calculation, does not have sufficient accuracy to achieve turbine design life goals. Can a time-accurate code do better? Will it yield results closer to engine experience? The heat transfer about turbine rotor blades is truly unsteady. This can be clearly seen in Fig. 8, which shows the time-resolved Nusselt No. measured about the midspan of a turbine rotor blade (Abhari *et al.*, 1992). Although the unsteadiness is most pronounced around the leading edge where the heat transfer modulation is close to 100%, it is still considerable (20~30%) at the trailing edge.

Figure 9 compares a time-accurate two-dimensional viscous calculation to the measurement. Qualitatively the comparison is quite good. We note that instantaneous heat transfer rate can be both much higher or lower than either the steady state laminar or turbulent boundary layer predicted for this geometry; this emphasizes the unsteady nature of the flow. Figure 10 shows sketches derived from the calculation of the unsteady shock system as it is convected by a heat transfer gauge location and on the crown of the suction surface. The three
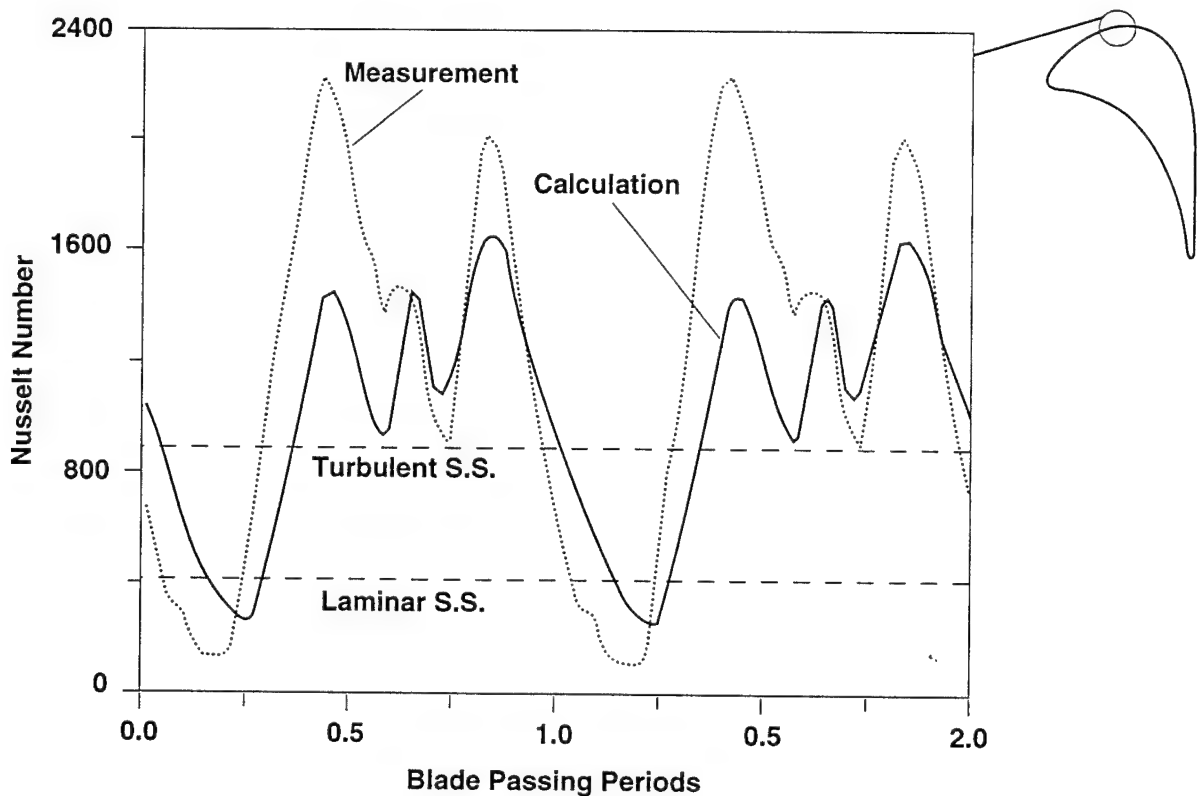

Fig. 9: Measured and calculated unsteady heat transfer on the rotor blade suction surface compared with laminar and turbulent steady state calculations for the same location.
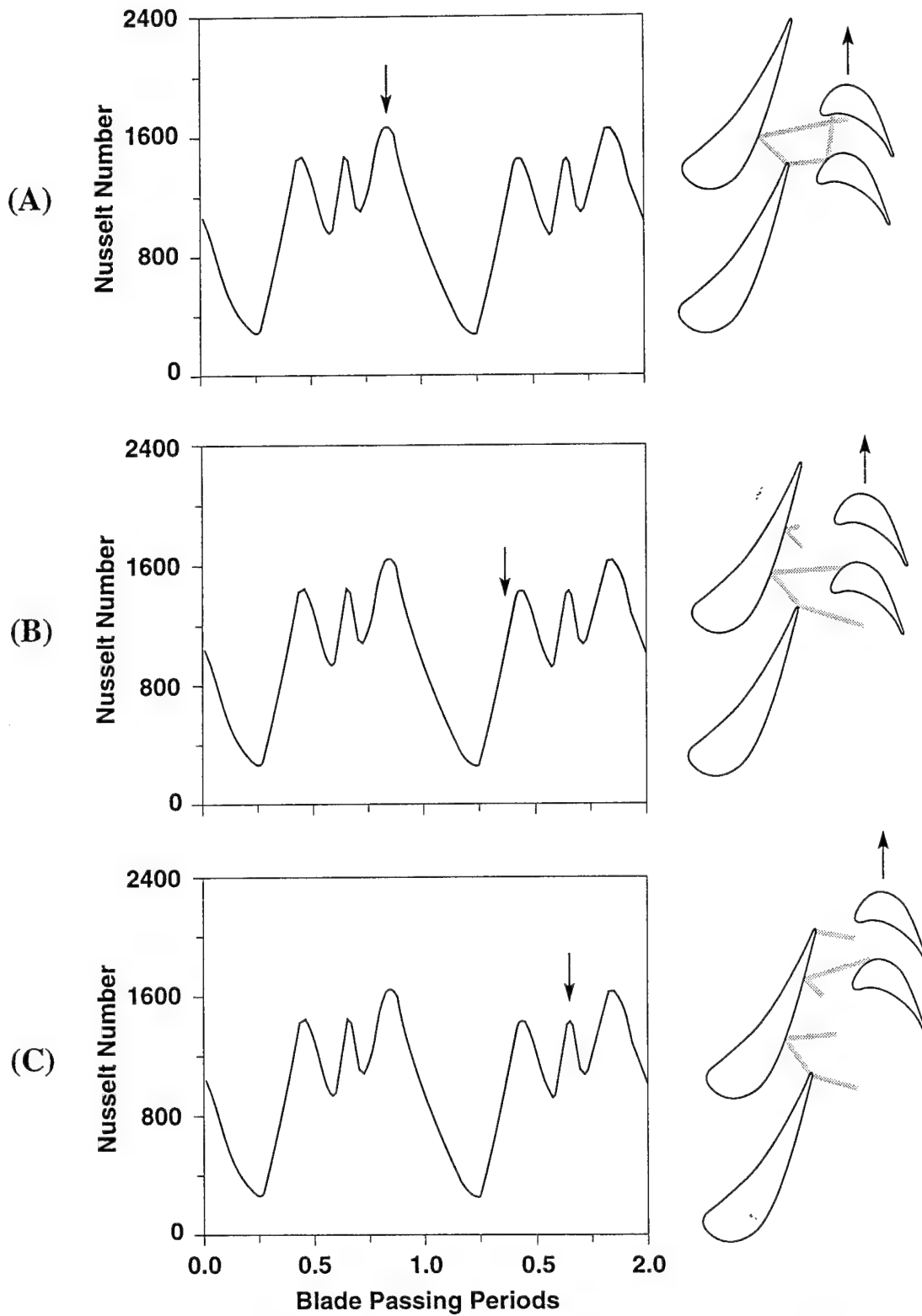
Fig. 10: The complex time-resolved measurement is clearly explained by the unsteady calculation animation as an interaction of the nozzle guide vane shock system with the rotor.

"bumps" in the data are clearly seen as due to (a) the primary nozzle guide vane (NGV) trailing edge shock, (b) the reflection of the trailing edge shock from the NGV suction surface, and (c) a secondary reflection of the trailing edge shock.

Although the CFD code predicts the qualitative structure of the data quite well, there are important quantitative differences, and it is these differences that are the most instructive on our weakness in the modelling and understanding. Most of the difference in the time mean heat flux between the calculation and measurement at the suction surface location shown in Fig. 9 is due to relatively low values which the calculation predicts for the three shock wave-induced heat transfer peaks. These peaks are due primarily to compressional heating of the boundary layer by the

On the pressure surface (Fig. 11), it is the boundary layer modelling which is the major moving shock waves (Rigby *et al.*, 1989). The unsteady viscous CFD code used for the prediction spreads the shock wave over several cells (as is common with such codes), reducing the pressure gradient and therefore the heat transfer (Abhari *et al.*, 1992). In this suction surface location then, the discrepancy between heat transfer calculation and measurement is due to the code's treatment of the inviscid flow, not that of the boundary layer.

On the pressure surface, the boundary layer modeling is a source of inaccuracy in this calculation. The thin shear layer code with the algebraic turbulence model used here does not couple the turbulence of the incoming periodic NGV wakes into rotor blade boundary layer. In the calculation, the boundary layer periodically relaminarizes (as evident from a time-resolved comparison of the ratio of laminar to turbulent viscosity). However the flow, as measured in the experiment, does not (Fig. 11). The result is a net underprediction of the mean heat transfer to the blade in this region.
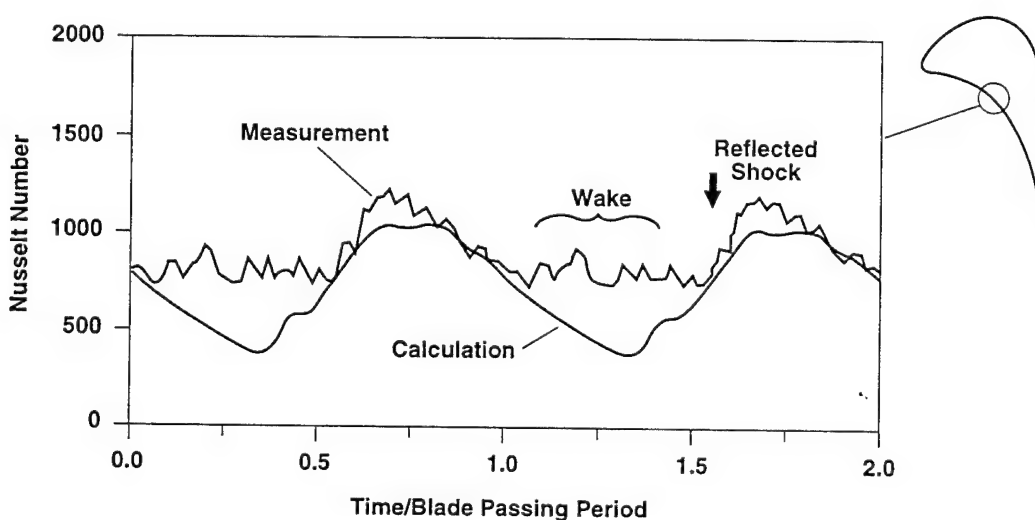

Fig. 11: Comparison of measured and calculated heat transfer on the rotor blade pressure surface. The improper calculation of the wake boundary layer interaction results in an erroneous low estimate of the average heat transfer at this location.

3.2.2 *Unsteadiness and Film Cooling*

From the above, it is clear that unsteadiness is of first order importance in uncooled turbine rotor blade heat transfer. There is much less information available on unsteady flows in film-cooled turbines. The data which does exist suggests that unsteadiness is even more important for film-cooled blades (Abhari and Epstein, 1994). Figure 12 illustrates heat transfer measurements on the same turbine geometry as discussed in the previous section, but with film-cooled nozzle guide vanes and rotor blades. Since both cooled and uncooled data is available, the cooling effectiveness [defined as 1 - (cooled Nusselt No.) / (uncooled Nusselt No.)] can be readily estimated. The data here show that the unsteady processes in the turbine modulate the cooling effectiveness by 100%, during a blade passing cycle, with values ranging from zero to twice the time average (Fig. 13). Furthermore, rectification processes generate a 12% difference between the average of the unsteady data and the steady state heat transfer.
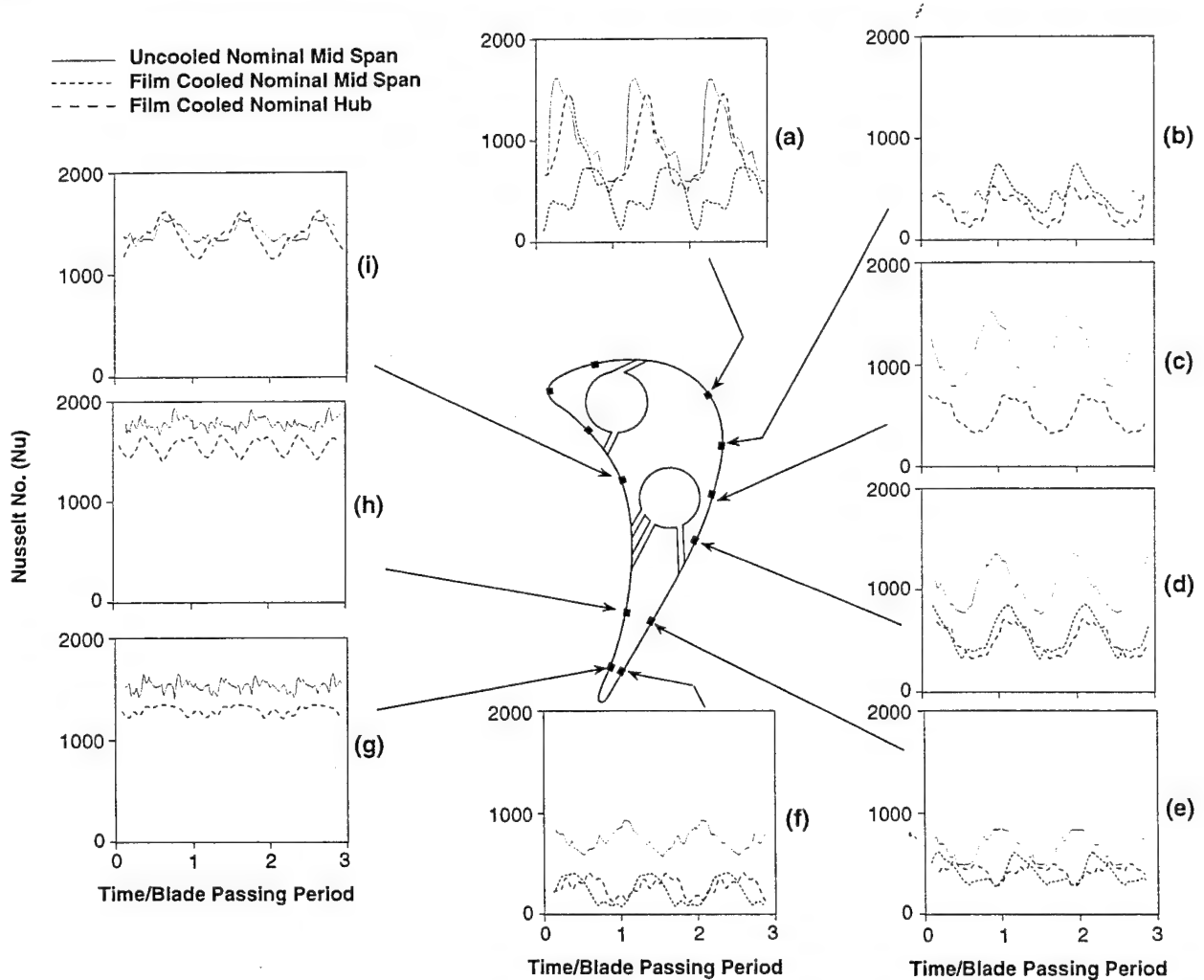

Fig. 12: Time-resolved rotor heat transfer measurements compared with those for an uncooled rotor.
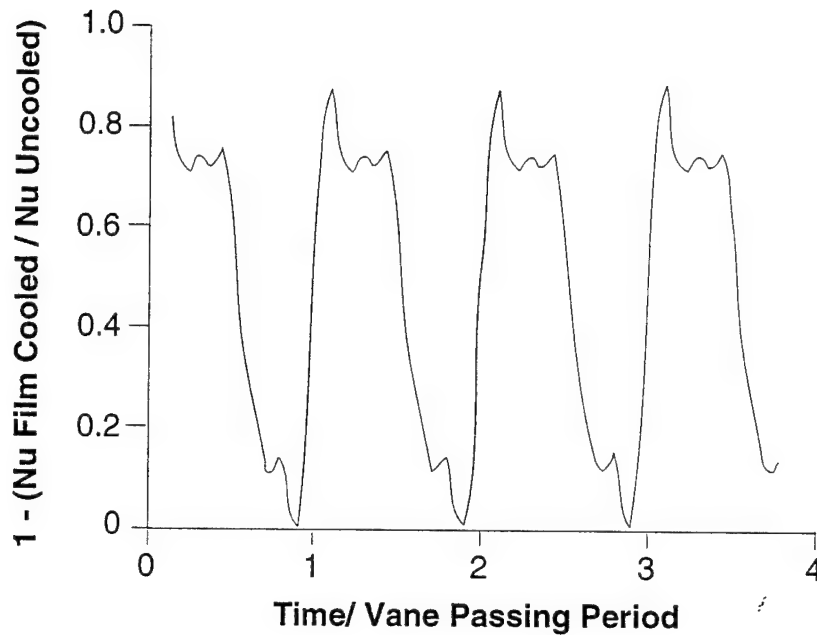
Fig. 13: Time-resolved isothermal effectiveness calculated from the midspan data on the crown of the rotor suction surface (Fig. 11).

One strong driver of this unsteady film cooling behavior is the unsteady static pressure distribution about the rotor blade generated by the inter-blade row interactions. Assuming that the pressure within the coolant supply plenum within the blade is constant, the time variation in the pressure ratio across the coolant holes as a rotor blade moves behind a vane can vary by as much as 40%, depending upon the position on the blade surface (Fig. 14). The resultant time variation in coolant momentum is sufficient to account for the observed fluctuation in rotor blade heat transfer, assuming quasi-steady behavior of the cooling films (*i.e.* the uncooled data is corrected for the time-varying coolant blowing to produce the film-cooled model predictions, which agree well with the film-cooled data in Fig. 15).

This high level of unsteadiness in film-cooled turbines has additional influence on the heat transfer process. One effect is that the unsteadily blown films such as those described above spread at different rates than a steady film does, given the same geometry (Bons *et al.*, 1995). This reduces the cooling effectiveness in a non-isotropic manner. Another influence of unsteadiness is that the very high freestream unsteadiness and turbulence levels inherent in turbines also alter the film spreading rate and cooling effectiveness (Bons *et al.*, 1994).
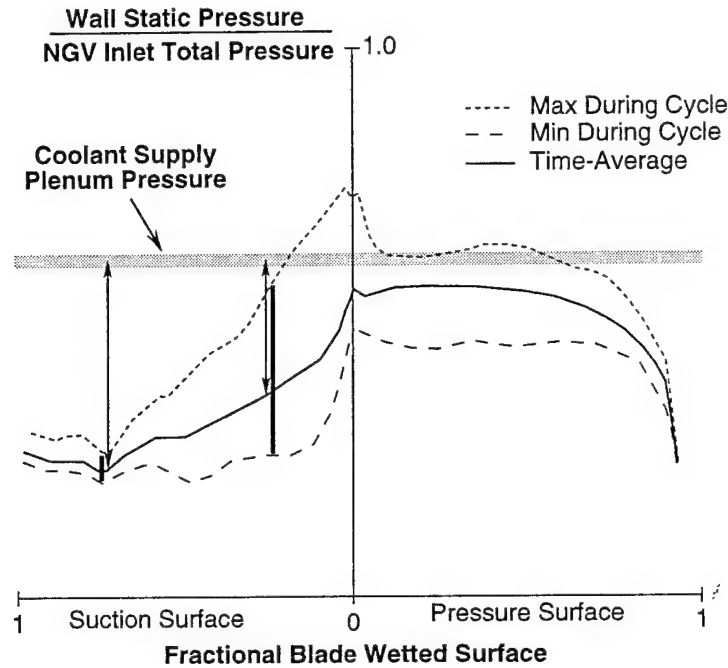
Fig. 14: Midspan static pressure distribution about a transonic turbine rotor blade. Unsteady vane blade interactions generate fluctuations about the average.
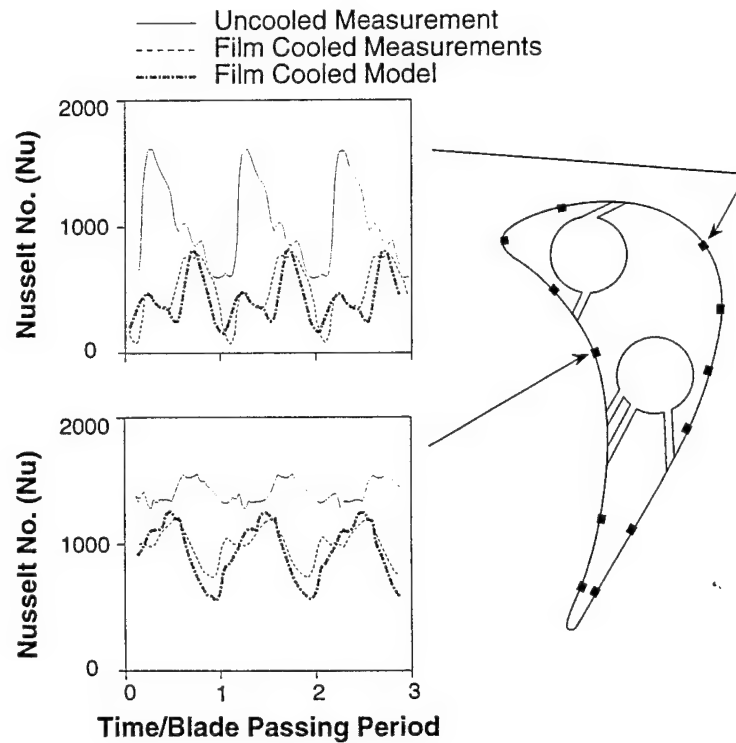

Fig. 15: A comparison of film cooled measurements and predictions made with a simple, unsteady blowing model.

3.2.3 *Summary Comments on Basic Understanding of High Pressure Turbine Aerodynamics and Heat Transfer*

The foregoing material makes four primary points. The first is that our basic understanding of heat transfer processes in turbines is insufficient to realize the aggressive performance, life, and cost goals needed to maintain aeropropulsion leadership into the next century. The second point is that fluid physics of high pressure turbines are primarily unsteady and must be studied as such. (We have probably gone as far as we can with the traditional simplifying assumption of primarily steady flow.) The third point is that the large scale, nominally inviscid flow features are at least as important to turbine heat transfer as the boundary layer details, and perhaps no better understood. The final point illustrated is that an interwoven set of high detailed measurements and calculations can be a powerful tool in elucidating the fluid physics in such complicated devices and highlighting shortfalls in our modelling of them.

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4. TASK II: FLOW CONTROL IN COMPRESSION SYSTEMS BY VISCOUS FLOW REMOVAL (J.L. Kerrebrock)

4.1 Motivation

Both the pressure ratio and the efficiency of compressors are ultimately limited by viscous flow phenomena that have their origin in thin shear layers on the surfaces of the blades, casings and hubs. The pressure ratio that can be developed by an optimized stage operating at some level of blade tip Mach number is limited by the diffusion that can be sustained in the blade passages without the development of massive separation from the suction surfaces of the blades, or from the hub and casing surfaces. The efficiency that can be realized is limited by the entropy rise suffered in these same shear layers, and in their mixing with the core flow. This research is directed at improving both the pressure ratio per stage and the efficiency of compressors, by the judicious removal of the viscous flow from the flow path in such ways as to reduce the tendency toward separation, and also to reduce the entropy rise suffered by the flow in the compression process, thus improving the efficiency.

The potential benefits of such changes are readily understood at the basic level. Increasing the pressure ratio per stage will reduce the number of stages required for a given overall pressure ratio, hence the size and weight of the compression system. There is also the possibility of making the flow in the compressor more predictable by minimizing the uncertainties due to viscous effects, leading to a more tractable design process. Removing the high entropy fluid developed in the boundary layers from the flow path, offers a reduction in compression work in succeeding stages, hence an improvement in efficiency. It also minimizes the entropy production due to mixing of low energy flow with the core flow.

In an overall engine system context, neither of these effects is necessarily a clear benefit. The effect on the engine depends on how the flow that is removed is managed. The removed flow must be incorporated into the overall engine flow in the most efficient way if the gains are not to be dissipated. As an example of such bleed flow management we may take the use of the bleed flow from supersonic inlets to cool the afterburner and nozzle of supersonic engines.

4.1.1 Research Focus

These comments serve to motivate the principal research thrusts of the program. They are:
1) Experimental and computational study of suction enhancement of the pressure ratio of high-performance compressor stages.
2) Theoretical and experimental study of the effects of suction on the efficiency of compressors.
3) Systems studies of the benefits to engine systems of the use of fluid extraction in the compressor.

4.2 Previous Work

The use of suction to enhance the work capability of compressor stages is not a new idea. It has been proposed many times, but there seem to have been few systematic experimental or analytical studies of its potential. Most of the work directed at boundary layer control has focused on slotted blades and on casing treatment. In the former the suction surface flow is reenergized by injection of fluid from the pressure side of the blade. In the latter, the casing layer is energized by mixing with the core flow in one way or another. The results of one rather extensive effort to improve the turning capacity of compressor stators by suction or blowing is summarized in Ref. [1]. In this work several stators with a variety of suction and blowing arrangements on the suction surfaces of the vanes, and some with suction on the hub and casing, were tested behind a high-performance guide vane and rotor. The stators were designed with very high Diffusion Factors, up to 0.75. The suction was implemented by means of slots in the otherwise normally contoured suction surface of the vanes. There is evidence, both experimental and computational, from flat plate studies that such slotting is not optimum, rather a raised trailing edge is desirable to avoid diffusion in the core flow due to the boundary layer removal. Nevertheless, the results were that fluid removal on the suction surface of the vanes gave significant benefits in stator performance. Blowing systematically degraded the performance.

We are aware of no systematic work in this area between the completion of that reported in Ref. [1] and the initiation of the present program in late 1992. There appear to have been no attempts to actually incorporate suction control in high performance compressors, in spite of the favorable results of Ref. [1].

The effort described here and proposed for extension has thus far had three principal components:
1) A study of the effect of boundary layer removal on compressor efficiency,
2) Experimental study in an existing stage, of the effects of fluid removal,
3) A design study of the increase in pressure ratio per stage that can be realized by using boundary layer suction, for a low-tip speed fan stage.

Each of these is complete or nearly so and the results will be summarized briefly below after a brief description of the proposed future work.

4.3 Summary of Status of Current Work

4.3.1 Thermodynamic Benefits of Viscous Flow Removal

One of the premises of this research program is that it can be beneficial to the overall performance of a compressor, or to an engine incorporating it, to remove the fluid influenced by viscous effects from the flow path, rather than mixing it with the inviscid flow and continuing to compress it. This is an unconventional approach; most other approaches to controlling separation

envision using blowing or slotted airfoil sections that energize the viscous-dominated flows to prevent separation. While this approach has much to recommend it, it leaves the high entropy fluid in the flow path, raising the average entropy (temperature) of the flow and increasing the compression work in successive stages of compression. The present approach, of removing the viscous-dominated flow, lowers the entropy of the residual flow and so minimizes the work in succeeding stages. The suction scheme is shown schematically in Fig. 1.
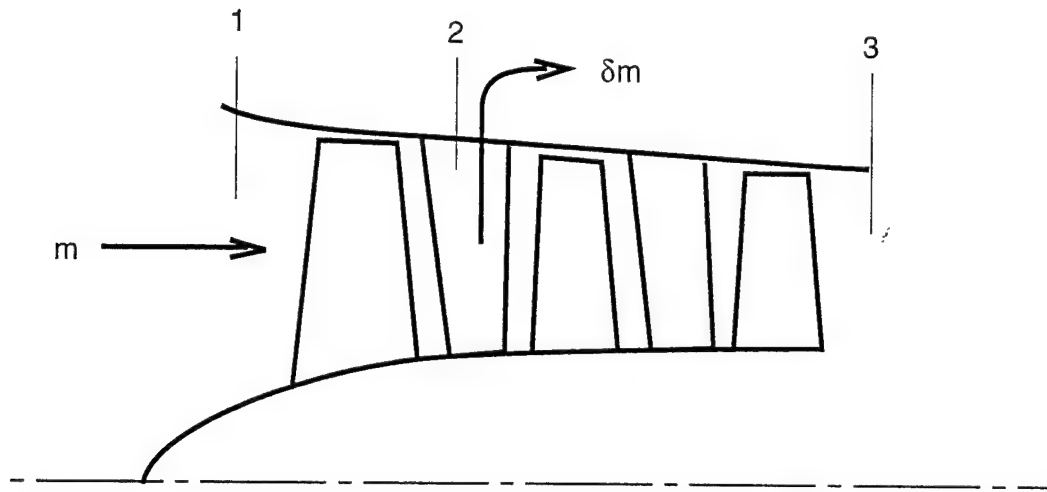

Fig. 1: Schematic of suction removal of high entropy flows.

At any particular blade row a small flow δm that has suffered an entropy rise due to viscous effects, is extracted from the flow path. The remainder of the flow $(m-\delta m)$ continues through the compressor. The thermodynamic effect is shown on a T-S diagram of the compression process on Fig. 2.

Point 1 is the inlet and 3 the discharge of the compressor. At some intermediate point 2 the suction is implemented. The solid line 1-2-3, indicates the compression process in the absence of suction. At point 2 the flow is divided into two portions, a small fraction that has undergone high viscous interaction and hence has a high entropy, indicated by condition 6, and the remainder with a low entropy, at condition 4. This would be done for example by a boundary layer scoop, as discussed later in this report. The high entropy fluid is extracted, and (for the purpose of this analysis) expanded back to the compressor entrance pressure at point 7 to recover as much work as possible from it, while the remainder continues in the compression process to point 5. In application the bleed air might be used for cooling of the engine hot section, for example, but rather than complicate the present analysis with such complex systems questions, the best possible situation from the viewpoint of efficiency is assumed.
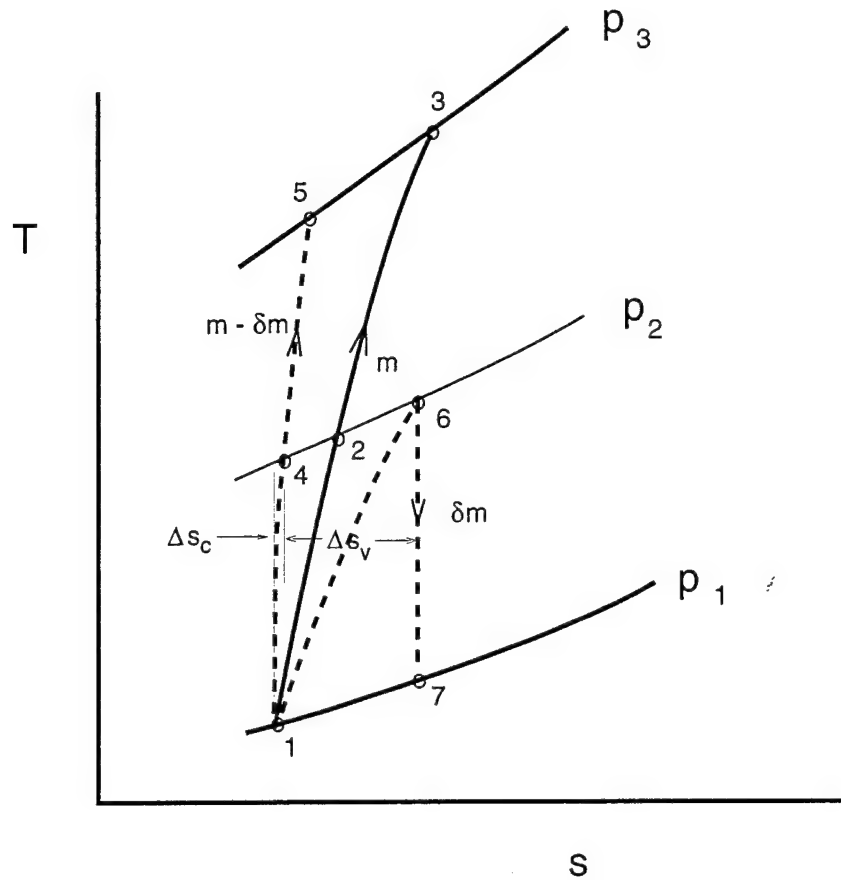
Fig. 2: Schematic of the suction scheme on T-S coordinates. p_1 is the inlet pressure to the compressor, p_3 the discharge pressure and p_2 the intermediate pressure at which the flow is segregated and the high entropy fluid is extracted.

The essential point here is that the main flow at point 5 ends up at a lower entropy (and temperature) than it would have in the absence of suction at point 3, so the work required for a given pressure ratio is lower and the resultant compressor efficiency is higher. Of course such suction can be applied repeatedly in each stage. Here we analyze the effect of suction at one location.

For a compressor <u>without bleed</u>, the work of compression can be written

$$W_{nb} = mc_p(T_3-T_1) = mc_pT_1\left[\left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\gamma\eta_p}} - 1\right]$$

where η_p is the polytropic efficiency. In the following argument we will assume as a first approximation that the polytropic efficiency is uniform for all compression processes. With bleed of amount δm at a point 2, the work is

$$W_b = mc_p(T_2 - T_1) + (m - \delta m)c_p(T_5 - T_4) - \delta mc_p(T_6 - T_7)$$

At the pressure p2, the flow is assumed to be divisible into a small portion δm that has suffered an entropy rise due to viscous effects, and the larger fraction $(m - \delta m)$ that is considered the core flow. We further assume that the separation is done in such a fashion that the pressure of the higher entropy fluid and the core flow are equal. Then the entropy of the extracted flow relative to that of the average flow at point 2 can be written:

$$\Delta s_v = c_p \ln\left(\frac{T_6}{T_2}\right)$$

The temperature of the core flow at 4 is related to the average temperature at 2 and that of the extracted flow by:

$$mT_2 = (m - \delta m)T_4 + \delta m T_6$$

The temperatures at the points 2, 5 and 7 are related to the compressor pressure ratio and the bleed pressure ratio by:

$$\frac{T_2}{T_1} = \left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma \eta_p}}$$

$$\frac{T_5}{T_4} = \left(\frac{p_3}{p_2}\right)^{\frac{\gamma-1}{\gamma \eta_p}}$$

$$\frac{T_6}{T_7} = \left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}}$$

Substituting these relations into the expression for the work with bleed, we have for the work per unit mass of flow delivered to the exit of the compressor,

$$\frac{W_b}{m - \delta m} = \left(\frac{m}{m - \delta m}\right)c_p T_1\left[\left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma \eta_p}} - 1\right] + c_p T_4\left[\left(\frac{p_3}{p_2}\right)^{\frac{\gamma-1}{\gamma \eta_p}} - 1\right] - \left(\frac{\delta m}{m - \delta m}\right)c_p T_6\left[1 - \left(\frac{p_1}{p_2}\right)^{\frac{\gamma-1}{\gamma}}\right]$$

Here the terms represent respectively the work done to bring the entire flow to the point 2, the work done to finish the compression of the main flow to point 5, and the work recovered by expansion of the extracted flow. Note that we have assumed this latter process to be ideal.

Eliminating T4 and T6 with the energy balance at point 2 and the definition of Δs_v, and assuming $(\delta m/m) \ll 1$, we find the difference of the work per unit of <u>delivered flow</u> with and without bleed to be

$$\frac{W_{nb}}{mc_p T_1} - \frac{W_b}{(m - \delta m)c_p T_1} = \left(\frac{\delta m}{m}\right)\left[1 + \left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\gamma \eta_p}}\left(e^{\frac{\Delta s_v}{c_p}} - 1\right) - e^{\frac{\Delta s_v}{c_p}}\left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}\left(\frac{1}{\eta_p} - 1\right)}\right]$$

This expression gives the reduction in work, per unit mass of delivered flow, as a result of the bleed. It is clear from its form that for a given entropy difference the gain is largest when the bleed occurs at a low pressure, i.e. near the front of the compressor, and when the pressure ratio is high. But it is easier to see the significance of the result if we divide it by the compression work without bleed, to get the fractional reduction in work. The final result is then

$$\frac{\frac{W_{nb}}{m} - \frac{W_b}{(m-\delta m)}}{\frac{W_{nb}}{m}} = \left(\frac{\delta m}{m}\right)\left\{ e^{\frac{\Delta s_v}{c_p}}\left[1 - \frac{\left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}\left(\frac{1}{\eta_p}-1\right)}-1}{\left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\eta_p}}-1}\right] - 1\right\}$$

If the quantity in curly brackets is positive, there is a net reduction in work for a given pressure ratio, hence a corresponding increase in efficiency. We see that the increase depends on the magnitude of the entropy excess in the bled flow, on the compressor pressure ratio, and on the pressure ratio at which the bleed occurs. By inspection we can see the following:

- The improvement in efficiency increases with increasing entropy excess. This implies that the gain will be largest for high Mach number blading.
- The improvement is larger for smaller bleed pressure ratios, i.e. it is most profitable to bleed the front stages of a compressor.

To assess the likely magnitude of $\Delta s_v/c_p$, consider a boundary layer-like flow over a compressor blade surface. Assume the static pressure in the core flow and at the surface are equal, and that the surface is adiabatic. Then the temperature at the surface is close to the stagnation temperature, and the entropy excess of the boundary layer fluid over the core fluid is

$$\frac{\Delta s_v}{c_p} = \ln\left(1 + \frac{\gamma-1}{2}M^2\right)$$

where M is the Mach number relative to the blade. Therefore the exponential coefficient containing the entropy rise is of the order of $1+ (\gamma-1)M^2/2$. Substituting this estimate in our general result, we find

$$\frac{\frac{W_{nb}}{m} - \frac{W_b}{(m-\delta m)}}{\frac{W_{nb}}{m}} = \left(\frac{\delta m}{m}\right)\left\{ \frac{\gamma-1}{2}M^2 - \left(1 + \frac{\gamma-1}{2}M^2\right)\left[\frac{\left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}\left(\frac{1}{\eta_p}-1\right)}-1}{\left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\eta_p}}-1}\right]\right\}$$

From this form of the result it is clear that the principle gains are to be realized for large relative Mach numbers, in the first stages of compressors. Indeed there is a Mach number below which

no gain can be had by bleeding. This is given by

$$\frac{\gamma-1}{2}M^2_{min} = \frac{\left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}\left(\frac{1}{\eta_p}-1\right)}-1}{\left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\eta_p}}-\left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}\left(\frac{1}{\eta_p}-1\right)}}$$

Figure 3 shows that for most pressure ratios and bleed pressure ratios, the minimum Mach number is between 0.3 and 0.5, so that most modern compressors can benefit from such bleed. Here the ratio of overall to bleed compression ratios is used as a parameter. As an example, for a compressor pressure ratio of 20 if this parameter is 2 then the pressure ratio at bleed is 10.

Returning now to the actual gain in efficiency to be expected from bleed, the quantity in the curly brackets is shown in Figs. 4 and 5. This is the fractional decrease in work per fraction of mass flow bled. From Fig. 4, which is drawn for a relative Mach number of 1, we see that bleed at almost any point in the compressor, will produce two tenths of a percent increase in efficiency per percent of mass bled. This is of course only true if the bled fluid has the entropy level associated in this case with stagnation from a Mach number of 1. The benefit is quite insensitive to both overall pressure ratio and the pressure ratio at which the bleed occurs, a somewhat surprising result.

The gain from bleed increases with the relative Mach number of the flow at the bleed point as Fig. 5, drawn for a Mach number of 1.5, exemplifies. In this case the gain is nearly one half a point of efficiency per percent of bleed flow.

Fig. 3: Minimum Mach number for which bleed increases the efficiency, as a function of the compressor pressure ratio and the ratio, of compressor pressure ratio to pressure ratio at the bleed point. (for η_p =0.90).

Fig. 4:   Fractional gain in efficiency per fractional bleed flow as a function of compressor pressure ratio and bleed pressure ratio, for relative Mach number of 1.

Fig. 5:   Fractional gain in efficiency per fractional bleed flow as a function of compressor pressure ratio and bleed pressure ratio, for relative Mach number of 1.5.

4.3.2 *Experimental Studies of the Effect of Suction on Compressor Flows*
4.3.2.1 *Tip Vortex Control by Suction*

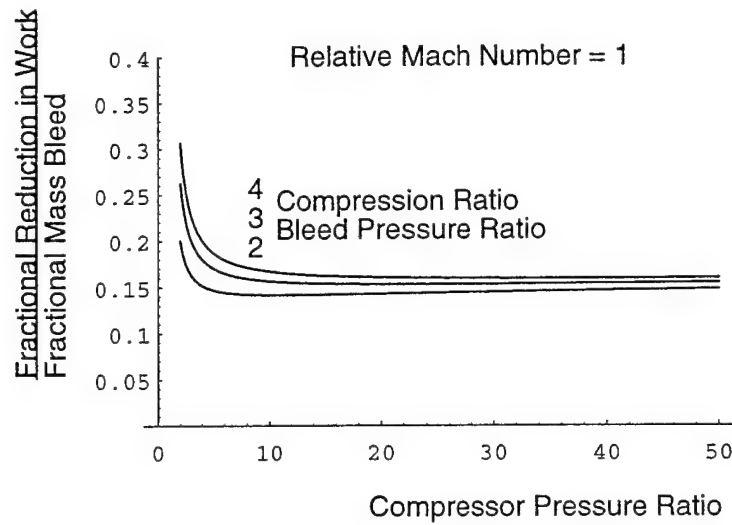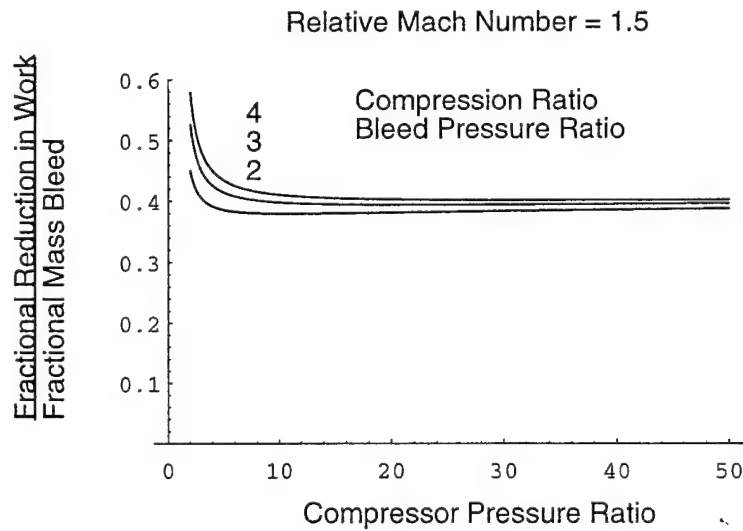The effect of suction through the blade on the tip vortex flow was studied experimentally and computationally by J.E. Dennis, and reported as his M.S. Thesis [4]. The original concept was that the jet emerging into suction side flow of blade passage, could be eliminated by such suction. Experimental and computational studies were undertaken in parallel. The computational work showed a small increase stage efficiency, but an increase in the clearance flow. The experiments confirmed the increase in the clearance flow. Thus the results of this investigation were negative, and this approach to flow control has been abandoned.

4.3.2.2 *Shock-Boundary Interaction Control by Suction*

This possibility is being studied by D. Reijnen as the subject of his doctoral research. The suction arrangement is shown schematically in Fig. 6. It has been implemented on five of the twenty-three rotor blades of the MIT transonic compressor stage. The experiments are in process at this writing. The results of this experiment will be used in the design of an enhanced stage.
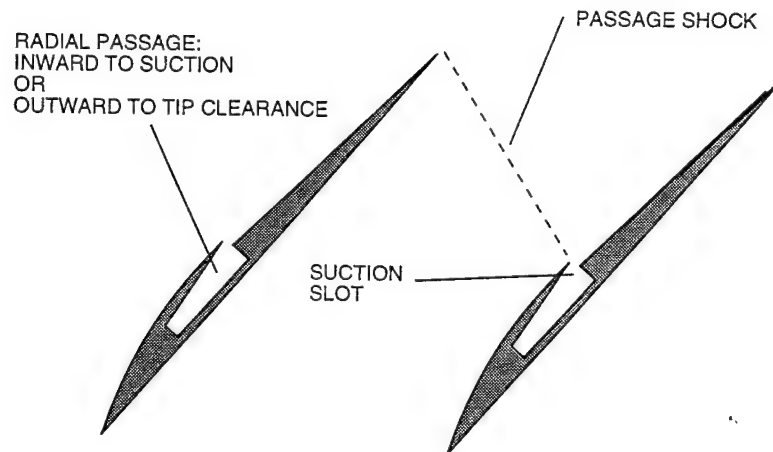
SUCTION AT SHOCK-IMPINGEMENT

Fig. 6: Schematic of suction arrangement at shock impingement location on suction surface.

4.3.2.3 Fan Stage Design Using Boundary Layer Suction

This aspect of the work was completed and reported as the SM thesis of L. Smilg [2]. The design was carried out using a combination of a classical streamline curvature program and the MISES code of reference [5]. The latter executes a momentum-integral boundary layer calculation in combination with a quasi-three dimensional inviscid flow calculation. The quasi-three dimensional feature accounts for streamline contraction in the radial direction.

The objective set for this design effort was a fan stage with a pressure ratio of approximately two at a tip Mach number of unity, the motivation being to enable a quiet fan stage with sufficient work to match the needs of advanced civil turbofans with bypass ratios in the neighborhood of ten.

Within the diffusion limits of conventional blading, the maximum pressure ratio available from such a stage is in the range of 1.5 to 1.6. The diffusion factors required for the target performance are shown in Fig. 7. They are well above the limit of 0.4 to 0.5 for normal blading.
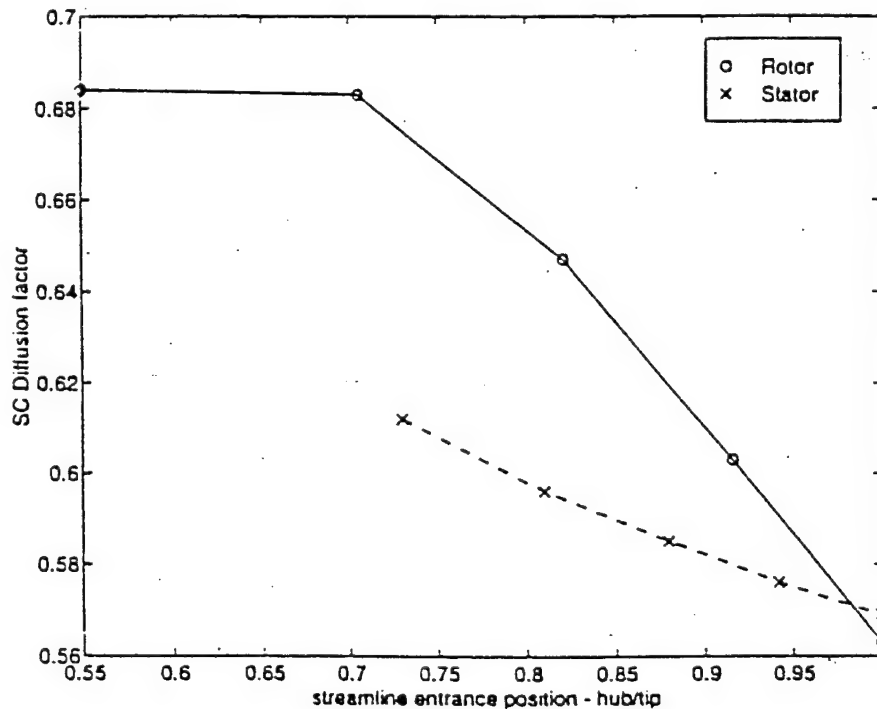

Fig. 7: Diffusion factors for the rotor and stator, as required to give the desired pressure ratio (2) at a tip Mach number of 1.0.

Suction was represented in the boundary layer computations by a step change in the momentum thickness of the boundary layer. In the free-stream flow it was represented by an overlap of the streamlines at the trailing edge (i.e. by a negative displacement thickness). After considerable iteration multiple-circular-arc blade sections and suction slot placements were identified for all radii of both the rotor and stator, such that the boundary layer behavior was predicted to be acceptable all the way to the trailing edge of the blades. Variations of momentum and displacement thickness for the mid span of the rotor are shown in Fig. 8.

Fig. 8: Displacement and momentum thicknesses for the rotor at mid-span.

The overall performance of this stage is summarized in Table 1, which lists the adiabatic and polytropic efficiencies at several radii and for the stage as a whole. It should be noted that this calculation presumes that the casing and hub flows will be well-behaved. After repeating it for a high-tip-speed inlet stage, we intend to test the stage selected for experimental evaluation by a full three-dimensional viscous calculation. This should reveal any problems at the end walls.

TABLE 1: STREAMLINE EFFICIENCY AND PRESSURE RATIO

Streamline	τ_c	η_c	π_c	η_{poly}
Hub	1.23	0.953	1.995	0.953
1/4 Span	1.23	0.969	2.017	0.969
1/2 Span	1.23	0.948	1.995	0.953
3/4 Span	1.23	0.934	1.976	0.940
Tip	1.23	0.919	1.957	0.926
Average	1.23	0.945	1.991	0.950

4.4 <u>Research Aspects to be Addressed</u>

Two principal thrusts are envisioned for the next phase of this research program. They are:

1) The design, construction and experimental evaluation of a compressor stage that takes advantage of suction control of the boundary layer flows,

2) A thorough systems evaluation of the use of suction-enhanced compressors in high-performance aircraft engines.

The first of these is a logical follow-on to the experimental work now nearing completion. It is being done on an existing compressor stage, and the suction is implemented on only a fraction of the blades on the rotor. The intent of this phase of the work has been to determine whether the effect of suction is that predicted by our computational modeling. The suction has been designed to remove the boundary layer flow just prior to shock impingement on the suction surface of the supersonic portion of the span of a transonic rotor with a pressure ratio of 1.6 at a tip Mach number of 1.2. Five of the twenty three rotor blades are equipped with suction scoops that exhaust through the hub of the rotor. Measurements of the rotor outflow will thus yield a direct comparison of the flow from blades with and without suction control of the boundary layer at this crucial location.

Because the majority of the blades have no boundary layer control, this experiment is not capable of evaluating the benefits of boundary layer control for overall stage(or rotor) performance. This must be done in a stage with boundary layer control at least on all rotor blades. In fact, since the enhanced rotor will produce a high level of outlet swirl, its companion stator will require boundary layer control as well. These points have been demonstrated computationally by Smilg [2] in the design of an enhanced fan stage. The construction and experimental evaluation of a stage that takes full advantage of the possibilities of suction is proposed as a major element of the continuing program.

Before undertaking the actual construction of the new stage the work of Smilg will be extended to a high-tip speed stage such as might be suitable as the first stage of a high performance core compressor. Then either the fan stage or this inlet stage will be selected for a full three-dimensional viscous analysis prior to the decision to build the stage.

After construction the stage will be evaluated in the MIT Blowdown Compressor Facility [3]. As part of the ongoing program, this facility has been improved by the addition of a fast-acting valve in place of the diaphragm that was originally used to initiate the flow. Provision has also been made for recycling the Freon working fluid in keeping with modern environmental concerns.

The second element of the ongoing program will be a thorough systems analysis of the incorporation of the enhanced compression systems in engines. This will include consideration of the use of the bled air for turbine and nozzle cooling and for bearing and seal flows as well as direct overboard dumping.

4.5 <u>References</u>

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5. TASK III: UNSTEADY PHENOMENA IN TURBOMACHINERY ENDWALL FLOWS (E.M. Greitzer, C.S. Tan)

5.1 Introduction

There are a number of unsteady effects inherent in turbomachinery operation, with a diversity of length and time scales. The length scales range over several orders of magnitude from the machine diameter to scales much smaller than a blade chord. Because time scales often (roughly) relate to length scales by convective velocity, the variation of the latter is also several orders of magnitude, from rotor revolution frequency to frequencies several times higher than blade passing.

The diversity of scale and frequency implies that many different unsteady flow phenomena occur in turbomachines, so that a prominent feature of these devices is the rich array of physical phenomena that one can encounter within a single machine. A critical item for engineering aspects which cannot be over-emphasized, however, is that *not all of the effects are important*, in the sense that not all of them need be taken into account in developing rational design procedures for multistage machines, nor are they all significant in terms of intrinsic scientific interest. A primary objective of this task can therefore be described as developing quantitative (i.e., predictive) understanding of the key issues which relate to unsteady flow effects in turbomachines. The aim is not only understanding but development of strategies for managing these effects. A second objective is to clarify, in a conceptual sense, the roles of the various unsteady fluid dynamic processes, with the sense of providing a "road map" to guide strategies for future research on issues of high scientific and technological payoff.

5.2 Background and Previous Work

5.2.1 *Dynamic Response of Multistage Compressors to Non-Uniform Inlet Conditions*

The work carried out during the past several years reflects both of the aspects described above. One part of the work has been completion of a phase of research which addresses unsteadiness with characteristic scale large compared to that of a blade chord (and hence a frequency comparable with that of the shaft revolution). This is the same frequency range as the fundamental aerodynamic instability in turbomachines, rotating stall, so that there is potential for strong interaction between the imposed unsteadiness and the "natural" instability and hence degradation of the stable flow range.

A detailed description of this interaction is given by Longley *et al.* (1994), but we can summarize the problem investigated and the main results.

The context of the problem is a two spool gas turbine engine, in which the presence of rotating stall in a upstream compressor imposes a rotating flow pattern, or distortion, on the
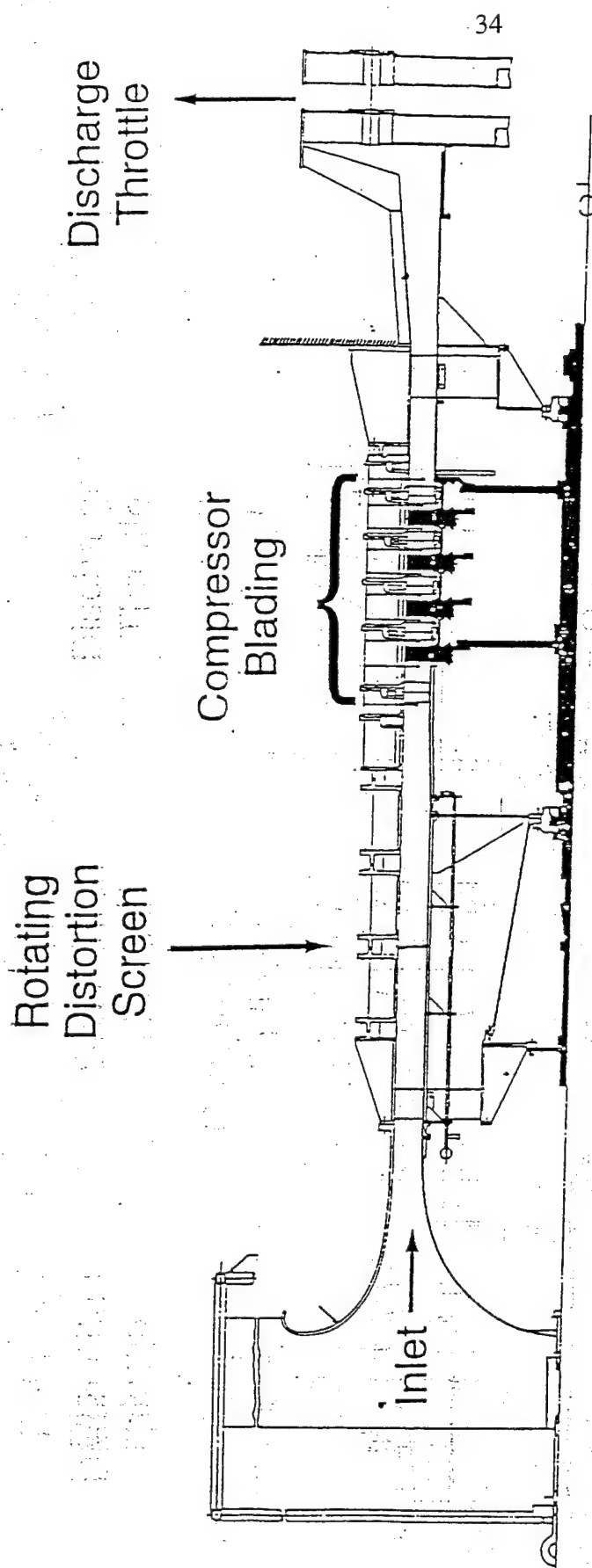
downstream compressor. This distortion affects the performance and, more importantly, the stable flow range. The goal was to extract from this complex dynamical system a simple, low order, description that would adequately capture the relevant phenomena. The modeling inherent in the research meant that the approach needed to include experiments, as well as analysis, so that the theoretical model could be assessed.

The theoretical model developed utilizes a two-dimensional (x,θ) unsteady description of the flow fields within the compressor and upstream and downstream. The assumption that the disturbances have long length scale compared to the blade pitch allows a simple description of the behavior of a multistage machine, in which the blade passages are modeled essentially as channels. The question continually asked in the model development was what degree of sophistication is needed to capture the important features of the flow; in spite of the approximate treatment adopted, it will be seen that the model does have this capability. The details of the theoretical description are discussed in the paper.

Experiments to assess the theory were carried out using the four-stage Large Scale Research Compressor in the Aerodynamics Research Laboratory at General Electric Aircraft Engines. The facility is shown in Fig. 1. In the experiments, the rotating distortion was simulated by a rotating screen upstream of the compressor. Overall (global) measurements were made of the multistage compressor responses, in terms of changes in stall onset, to speed and direction of rotation of an inlet distortion. Detailed (local) measurements were also made to identify the nature of the unsteady evolution from essentially axisymmetric flow into fully developed rotating stall.

For distortions rotating in the same direction as the compressor rotor, the experiments showed that the compressors exhibited significant loss in stable flow range and that they could be divided into two groups according to their response. The first group exhibited a single peak in stall margin degradation when the distortion speed corresponded to roughly 50% of rotor speed. For this group the theory gave a good description of the loss in stability as shown in Fig. 2. In the figure, the vertical axis is the compressor flow coefficient at the stability point, and the horizontal axis is the rotation rate of the upstream disturbance (i.e., of the upstream screen) normalized with respect to the rotor shaft speed. The theory and the experimental results are both shown; the different sets of points correspond to data at different shaft speeds. It can be seen that at a screen rotational speed near one-half shaft speed, the compressor has suffered a marked decrease in stability, losing almost 90% of the stable flow range it had with a stationary (non-rotating) distortion.

We take this first-of-a-kind comparison as showing that the theoretical description, although simplified, does capture essential fluid dynamic features of the phenomena. Using the theory, the compressor dynamical response can be interpreted as a nonlinear resonance between the forcing due to the distortion and the disturbance structure in the compressor. The nonlinearity is critical, because for a linear system forcing can have no effect on stability. The theory also

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Rotating Distortion Screen

Compressor Blading

Discharge Throttle

Inlet

$rr = 0.85, \quad r_t = 1.52 \text{ m } (5.0 \text{ ft})$

$U_t = 40 \text{m/s (at 500rpm)}$

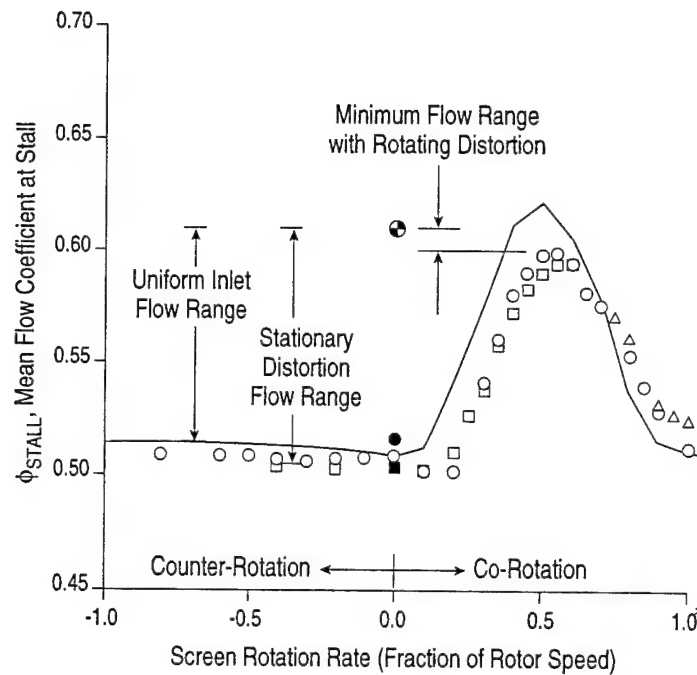Fig. 1: General Electric Low Speed Research Compressor.

Fig. 2: Compressor response to rotating inlet distortion: flow coefficient at stall versus distortion rotation rate.

shows features of the disturbance behavior which are representative of a class of wave motions in non-homogeneous media. There is growth of the wave over one part of the compressor annulus, and decay over another. The conditions at which the growth and decay are balanced is the neutral stability point. The theory also allows connection of the wave growth with the local performance of the compressor. The parts of the annulus where the wave growth occurs are regions in which the compressor is locally stalled and thus feeds energy into the disturbances. In the regions of wave decay the compressor is unstalled and there is damping of disturbances.

The second group of compressors examined showed a different behavior, which is seen in Fig. 3. These compressors exhibited two peaks in stall margin degradation corresponding to distortion rotation speeds of approximately 25-35% and 70-75% of rotor speed. The result that multistage compressors can have more than one resonant peak is also believed to be new.

The detailed measurements gave the clues to the reason for the two types of behavior. These measurements showed that the unsteady response could be directly linked to differences between the stall processes for the two groups of compressors. For the first, stall inception is essentially two-dimensional, in accord with the assumptions of the theory. In the second, the stall process has a strongly three-dimensional component, so that the behavior would thus not be expected to be well captured by the (two-dimensional) model.
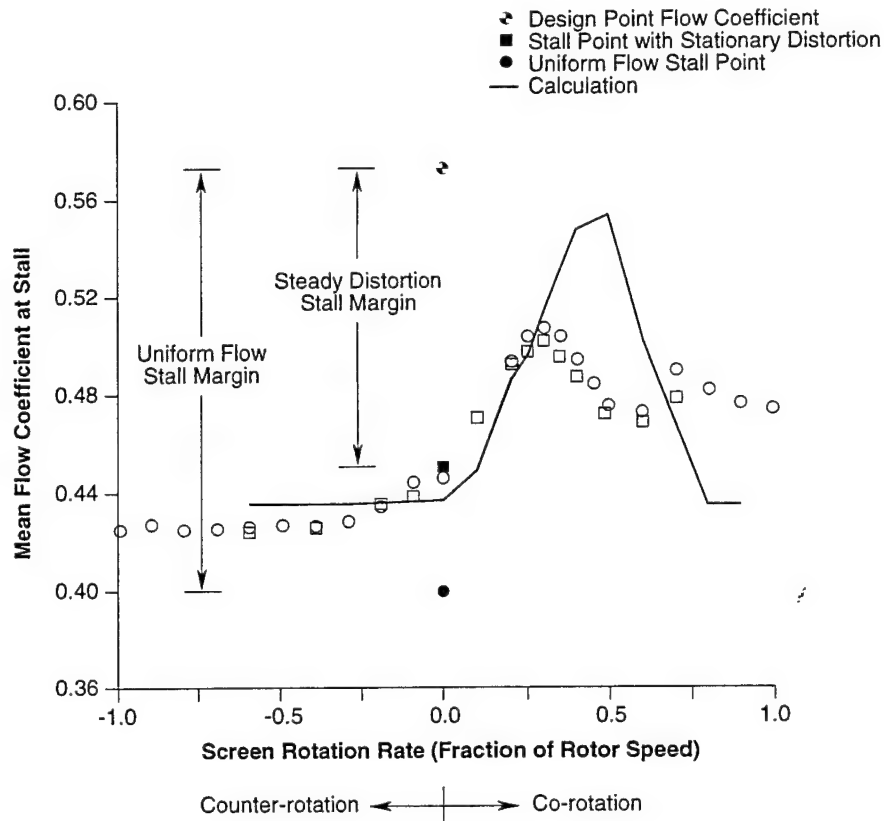
Fig. 3: Compressor response with three-dimensional stalling behavior.

From an engineering point of view the results show clearly that compressor stability margin depends on speed and direction of the rotating distortion. For distortions that rotated in a direction counter to that of the compressor, all the machines tested shown minimal loss of stability margin. These results imply that counter-rotation of the fan and core compressors, or of low pressure and high pressure compressors, could be a worthwhile design choice.

Finally, the work described represented a collaborative effort between the MIT Gas Turbine Laboratory and the Aerodynamics Research Laboratory at General Electric Aircraft Engines, under the direction of Dr. D. C. Wisler. The experimental work was conducted at the GE facility by MIT students, working with GE personnel, and we regard it as an example, in *both process and content*, of successful collaboration between university and industry. In addition to supplying partial funding, GE personnel have participated materially in the design of the experiment and in the data acquisition and analysis. The students have thus been given the opportunity to work with first rate technical colleagues on a problem of interest to industry as well as to university. In addition, the experimental facilities, which go beyond those that a university can muster, make use of turbomachinery with advanced aerodynamic design, so that the results are representative of the

phenomena encountered in machines that are of high practical interest. Finally, this type of interaction provides a strong conduit for the transition of ideas, from academia to industry as well as from industry to university.

5.2.2 *Effects of Asymmetric Tip Clearance on Compressor Stability*

The theory that has been developed has also recently been extended to examine another, quite different problem, which is also characterized by length scales large compared to a blade pitch. This is the overall behavior of a multistage compressor with asymmetric tip clearance (Graf, 1994). A situation of interest to the engine community is the influence of such asymmetry on compressor stability. The problem is structural in origin (it arises, for example, from eccentricity or casing deformation), and shows one feature of the coupling between aerodynamic and mechanical phenomena in a gas turbine engine.

Using an extension of the theory developed for inlet distortion, we have analyzed the behavior resulting from a clearance asymmetry due to casing shape. The driving phenomenon is that a clearance that varies around the circumference causes the local compressor pumping (the pressure rise), and the local mass flow to vary around the circumference. An example of the resulting envelope of local compressor performance (*i.e.* the locus of compressor performance at different positions around the circumference) is shown in Fig. 4. The heavy solid line in the figure is the curve of compressor pressure rise versus flow coefficient for a nominal, and circumferentially uniform, clearance. The two dashed curves correspond to compressor performance at reduced and increased, but still circumferentially uniform, clearances. The solid oval line represents the locus of local operating points of the compressor for the asymmetric clearance, *i.e.* the local value of pressure rise and flow which the compressor produces at different positions round the circumference. For a given geometry, knowledge of this steady nonuniform flow is needed for computation of the aerodynamic stability for the machine.

A central result of the analysis was the change in stability as a function of clearance asymmetry amplitude and compressor characteristic (pumping curve) shape. Figure 5 shows numerical results (decrease in peak pressure rise and shift to higher flow for instability onset) for a three-stage compressor with a sinusoidal variation in clearance. There is a substantial effect on stability and peak pressure rise.

An operational question is whether one can regard the loss in performance as just an effect of average clearance, or whether one needs to consider the unsteady behavior. The answer can be inferred by examining the peak pressure rise and loss in stability. Peak pressure rise was found to correspond roughly to that for the maximum tip clearance, i.e., worse that would be estimated using the average clearance. The loss in stability was also worse than one would find based on the average clearance. Both of these results, which, as far as we are aware are the first predictions of
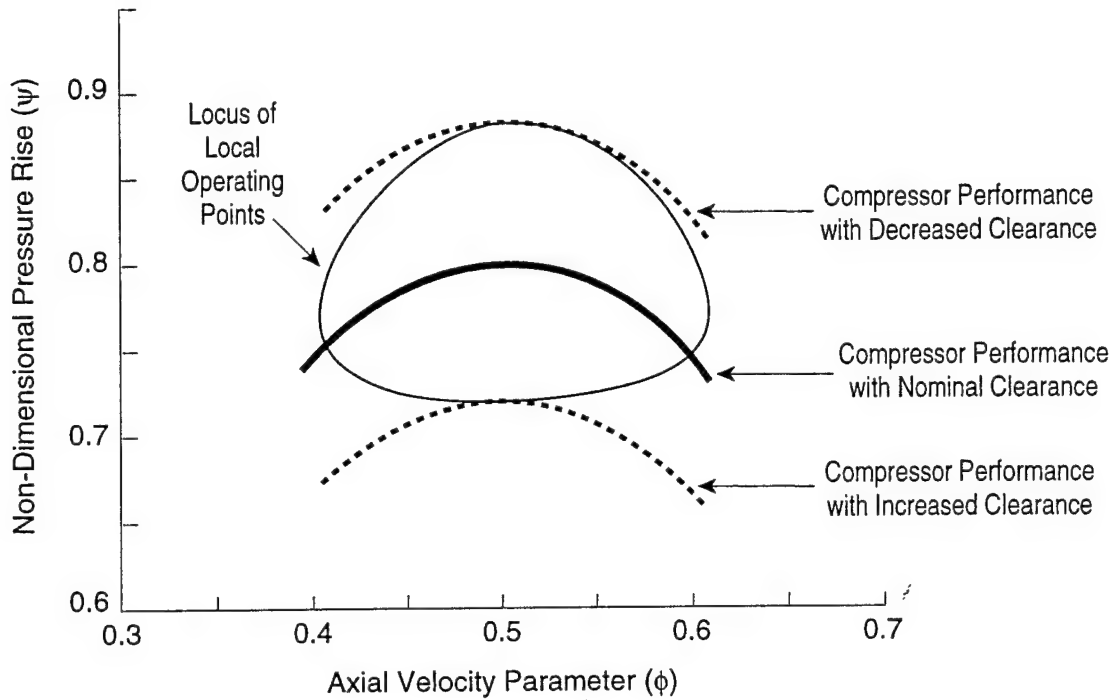
Fig. 4: Local compressor operating point around circumference with 2% chord sinusoidal tip clearance variation.
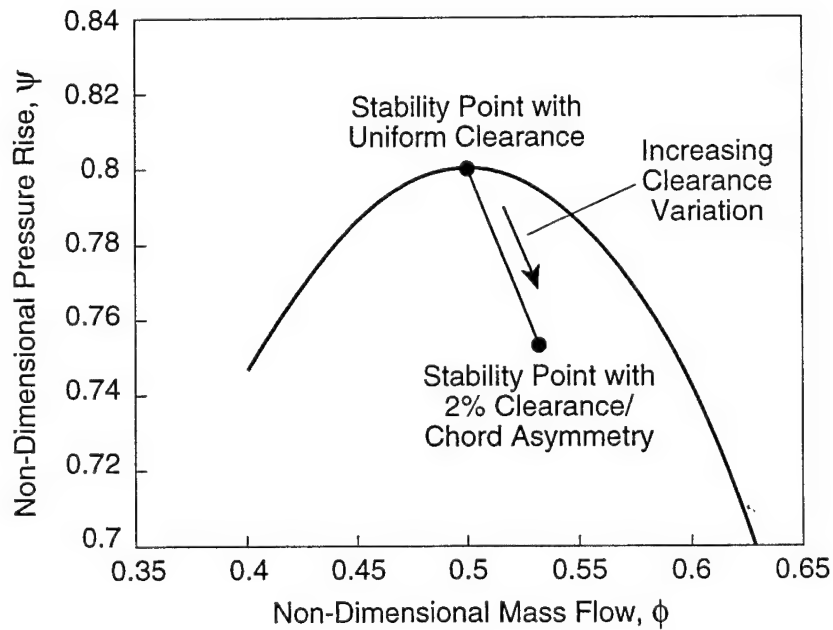

Fig. 5: Change in neutral stability point for sinusoidal clearance variation.

the effect of asymmetric tip clearance on stability, imply that an averaged view is not suitable and that description of the unsteady flow is needed to address the problem.

Other aspects of clearance asymmetry were also examined. The details are given in the referenced report, but some conclusions are presented here: (1) Increasing the magnitude of the clearance asymmetry decreases the stalling pressure rise and can increase the stalling flow coefficient. The loss in pressure rise capability scales roughly with the maximum of the clearance asymmetry. (2) Compressors which produce characteristics that have high peak pressure rise, narrow width, and steep fall-offs in pressure rise as a function of mass flow are more sensitive to clearance asymmetry than those with flatter characteristics. Figure 6, which shows the decrease in peak pressure for compressors with a nominal pressure rise and with pressure rises two and three times nominal, is an illustration of this trend. (3) The effect of clearance asymmetry on stall margin decreases as the dominant spatial harmonic which characterizes the asymmetry increases; clearance asymmetries with a single "lobe" around the circumference produce the greatest loss in stall margin. (4) For a given compressor, sensitivity to clearance asymmetry is not only a function of the compressor parameters, but is also related to the compression system dynamic behavior. In particular, because of the nonuniform background flow created by the tip clearance asymmetry, there can be an interaction between the disturbances which are local to the compressor and overall system type disturbances.

The present status is that experiments have been designed based on the theoretical analysis and are being carried out at General Electric.
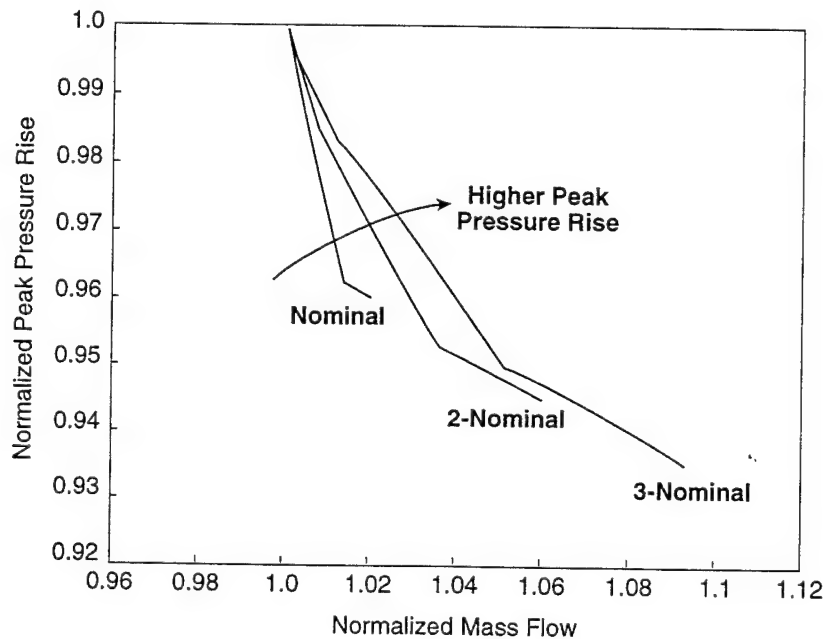

Fig. 6: Effect of pressure rise (number of stages) on decrease in stability: sinusoidal clearance variation.

5.2.3 *Some Perspective on the Impact of Unsteady Flow on Turbomachine Performance*

A second thrust of the work carried out concerns the more general topic of sorting out dominant unsteady phenomena in the multistage environment. An objective is to define key experiments and computations which will enable quantification of the role of specific unsteady flow phenomena on turbomachine aerodynamic and aeromechanical (*e.g.* high cycle fatigue) behavior. Workshops on this topic were held at NASA in June 1992 (Adamczyk *et al.*, 1992) and at Purdue in October 1993, the latter sponsored by AFOSR (Fant and Murthy, 1993). Our views of the topic, combined with results and conclusions extracted from the two meetings, have been incorporated into a paper which gives an overview of the field. The paper, "Unsteady Flow in Turbomachines, Where's the Beef?" is included as an Appendix to this document. It was presented as the keynote lecture in a four-session symposium on "Unsteady Flows in Aeropropulsion" at the ASME Winter Annual Meeting in November 1994. Co-authors of the paper (with Greitzer and Tan) were Dr. D.C. Wisler (GE), and Dr. A.J. Strazisar and Dr. J.J. Adamczyk (NASA Lewis Research Center). The paper thus represented a combined university/industry/ government viewpoint on the topic.

5.3 **References**

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6. COMMENTS ON THE MULTI-INVESTIGATOR FORMAT

We regard the multi-investigator format proposed here as one of the strengths of the technical as well as programmatic aspects of the research. The reasons behind this view are tied to the nature of the issues addressed. The problems investigated are complex, generally involving several different physical phenomena. As stated in the previous section, they are thus often situations in which the "tried and true" approximations that have served well in other areas of fluid dynamics are not adequate. One ready example is the effect of turbomachinery endwall flow on blade row performance. Analysis of this phenomena has been carried out from the viewpoint of analogy with channel diffusers, from a secondary flow perspective, from a lifting line vortex theory, and from integral and differential treatments of three-dimensional boundary layers. However, the endwall flow is truly three-dimensional, not "secondary", and the above approximations have not been adequate to deal with the problem of interest here.

This is only one example, but all of the tasks share the aspect that they are highly coupled, in terms of different fluid mechanic features, with complex structure. It has been our experience that it is extremely useful to be able to interact with other researchers when tackling problems of this type. In particular, the application of different views and perspectives, as well as even the general establishing of a line of reasoning becomes more critical when one is tying to assess which pieces of the problem are important and which can be neglected. A multi-investigator program can provide the avenue for fostering such interaction. This is true not only in faculty-to-faculty contact, but also for faculty-student and, possibly more important, student to student interaction. Indeed we have found that this last is a powerful means for our students to move themselves up the learning curve on different problems. In this context, the availability of relevant analytical and computational tools, as well as of data acquisition procedures, provides considerable leverage for the capabilities of individual students.

There are also several benefits as far as programmatic aspects are concerned. A view can be taken across the whole program so that, to some extent, moneys can be shifted internally to make the best use of resources, for example to put increased focus on new phenomena. It should be noted that to have this type of flexibility there must be close interaction, in a cooperative sense between the different investigators, including a strong sense of trust; increased effort for one project implies decreased support for another. This type of relationship, and the desire for such flexibility, exists in the present program and is fully subscribed to by the different investigators; it has been built up over a long period of time, and is one of the key features of the program.

In summary, we regard the synergism that results from interaction between investigators as a key aspect in the ability to attack problems that are not only intellectually challenging but of high technological interest. The multi-investigator format of the research program adds value in fostering this interaction.

Two other aspects of the research deserve mention. One is the existence of a "critical mass" of interacting faculty, staff, and students who are keenly interested in the problems to be address. The synergy resulting from such interaction implies a high degree of leverage for the research projects, which becomes more important as the complexity of the problems addressed increases. The second feature is the strong connection with industry and government laboratories. This has a number of benefits, concerning research as well as training graduate students. It gives access to an array of excellent experimental facilities, many beyond the scope of an academic institution, which embody features characteristic of advanced design. It also provides contact with real world problems. It gives opportunity for the type of team interactions that are becoming more necessary in aerospace technology. Finally, close contact with industry provides increased ability for dissemination of ideas and technology transition.

7. PERSONNEL WORKING ON THIS PROJECT

<u>Faculty and Staff</u>

    Professor A. H. Epstein

    Professor E, M. Greitzer

    Professor J. L. Kerrebrock

    Dr. C. S. Tan

<u>Graduate Students</u>

    C. Brown

    J. Dennis

    M. B. Graf (AFRAPT Student)

    H. Park

    D. Reinjen

    T. Shang

    L. Smilg

    D. Sujudi

    T. Wong

    W. Ziminsky (ASSERT Student)

8. PUBLICATIONS

Longley, J. P., Shin, H.-W., Plumley, R. E., Silkowski, P. D., Day, I. J., Greitzer, E. M., Tan, C. S., Wisler, D. C., "Effects of Rotating Inlet Distortion on Multistage Compressor Stability", ASME Paper 94-GT-220, 1994; to be published in *ASME J. Turbomachinery.*

Greitzer, E.M., Tan, C.S., Wisler, D.C., Adamczyk, J.J., Strazisar, A.J., "Unsteady Flow in Turbomachines: Where's the Beef?", in *Unsteady Flows in Aeropropulsion*, ASME Winter Annual Meeting, November 1994.

Shang, T., Guenette, G.R., Epstein, A.H., Saxer, A.P., "Influence of Inlet Temperature Distortion on Surface Heat Transfer of a Transonic Turbine Rotor", AIAA Paper AIAA-95-3042, AIAA Paper AIAA-95-3042, presented at the 31st AIAA/ASME/ SAE/ASEE Joint Propulsion Conference and Exhibit, San Diego, CA, July 1995.

9. TECHNOLOGY INTERACTIONS AND TRANSITIONS

Interactions and transitions took several forms:

a) General Electric provided partial support for the work on unsteady flow in compressors. This took the form not only of financial support, but also of engineering help, the furnishing of time on their research compressor, and, in general, collaboration in the overall research effort.

b) Mr. M. Graf carried out the computations for his Ph.D. program at Pratt&Whitney, working with both designers and method developers there.

10. NEW DISCOVERIES, INVENTIONS, OR PATENTS

None

UNSTEADY FLOW IN TURBOMACHINES: WHERE'S THE BEEF?

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ABSTRACT

Selected examples of unsteady flow in turbomachines are discussed from the point of view of their impact on aeroengine design. This vantage point serves as a focus for assessment of the "importance" of various unsteady flow processes. It is shown that for some of these processes there are clear causal links between unsteady phenomena and so-called steady-state performance measures. For others, however, the links, as well as the overall impact, are still to be established. Suggestions are given for areas in which research on unsteady flow can have a significant influence on design procedures as well as for ways in which to better define the topics that are most fruitful to pursue.

1. INTRODUCTION

Over the past several decades there has been a great deal of research on unsteady flow in turbomachines. The context in which the work has arisen, as well as the connection between the results of the research and the needs of the turbomachine designer, have varied over a wide spectrum. At one end is work on unsteady flow which is driven by engine operational issues, for example the structural problems described by an Air Force Scientific Advisory Board Ad Hoc Committee (1992), or the aerodynamic instabilities (surge and rotating stall) discussed by Day (1994). Problems such as these have a clear "traceability" between unsteady flow phenomena and operational limits. At the other end of the spectrum is work which is driven more by interest in the (often fascinating) aspects of unsteady fluid motions rather than by a specific goal tied to machine performance. A large gap lies between propo-

nents of work arising in each of these contexts who are often uneasy with one another and sometimes not willing or able to communicate.

The present paper is an attempt to provide some meeting ground for the diversity of interests in unsteady flow by suggesting one perspective from which to view unsteady flow phenomena, namely from the role of its impact on the design of high performance aeroengine turbomachinery. The approach taken can be stated as the following challenge to the worker on unsteady flow: From the point of view of the turbomachinery designer, unsteady flows are only important if they have a measurable impact on the machine performance, and knowledge about them enables clear design choices. It is with this guideline in mind that we will survey some of the issues that arise in this field.

1.1 The Current Environment for Research in Turbomachinery

The overall perspective from which the discussion of unsteady flow is to be phrased is given above, but there is another current factor in turbomachinery technology development that should also be noted. The large changes taking place in the aeroengine industry, driven by new business attitudes and practices, have had a major effect on money, people, type of technology funded, time, and attitudes. Support for enabling technology has been greatly reduced, and even support for critical technology (the technology you must provide to sell your product or fix a field problem) is critically scrutinized. There is little or no "technology for technology's sake" any more. Along with changes in support, the time available for completion of

projects has reduced. For example, time from concept go-ahead to certification in some cases has been reduced by over fifty percent. This, in turn, means that the time available for design is reduced. All of the above put a strong pressure for justification of any research. It is with the aim of not only showing what is needed for justification but also providing clear links from research issues to design guidelines in a number of topics that the paper has been written.

2. THEME AND SCOPE OF THE PAPER

We stress at the outset that we are in no way stating that unsteady flow research which is not targeted at a specific design problem cannot be both useful and of interest to the technical community. Rather, we start the discussion with the empirical observation that the most effective manner in which to convince designers that unsteady effects are of interest and should be factored into the design process is to show them a tangible benefit which is associated with paying attention to the effect. (The term "design process" is understood to include not only the design point but also the performance at the off-design conditions important to the flight envelope.) The set of consequences that stem from this observation can be regarded as a filter through which unsteady flow research is sampled. Use of such a filter leads directly to figures of merit that one should employ to assess the "importance" of unsteady flow. These are the turbomachinery and gas turbine engine performance measures that are used by designers: efficiency or thrust specific fuel consumption, aerodynamically stable flow range, adequate operating regime without undue aeromechanical excitation, heat transfer limitations and durability, reliability and ease of maintenance, and environmental considerations.

Use of this type of filter also leads to questioning a concept which is often stated as dogma, namely that a detailed understanding of, and full accounting for, all features of a complex flow will necessarily lead to better designs and major advancements. Again, this should not be misunderstood. Understanding and accounting for complex flow features is clearly important to the design process. What is stated is that it does not follow that if we had such a detailed understanding of and accounting for all unsteady phenomena in a newly-developed design system we would necessarily produce a better design and have major advancement.

One reason for this is that designers have often been able to engineer their way around things they do not understand fully. Indeed in aircraft engines, it is no exaggeration to say that in almost all aspects the device existed before the detailed understanding. For example, compressor polytropic efficiency can be 92% or higher. Will a

very detailed understanding of wake/boundary layer unsteady interaction bring about a major advancement in efficiency? A great number of empirical and computer parametric studies on airfoil shapes have been done to optimize geometry, and there have been outstanding successes. Can understanding more of the details of the unsteady flow phenomena help us to improve the design process? The answer to this question is not clear. On the other hand there are areas (such as aeromechanical interaction) in which unsteady flow effects are vital to the health of the product and in which it appears that further research and increased understanding can have a strong impact on the design process.

The content of the paper is as follows. In the next section we give a preface to the examples to be shown, followed by illustrations of the effects of unsteady flow processes on peak efficiency, heat transfer, aeromechanical response and aerodynamic instability. We then present one means of characterizing these diverse phenomena and use this as a lead-in to discussion of specific areas that are of interest for research. Comments are also made concerning questions and issues that should be addressed with respect to selection of future research topics in unsteady flow.

3. NATURE OF THE EXAMPLES OF UNSTEADY FLOW PHENOMENA TO BE SHOWN

The set of objectives that turbomachinery has to meet is a diverse one, and it is to be expected that phenomena that are inherently unsteady will be at the heart of some of these. As we examine some of the examples that illustrate the behavior of the various figures of merit listed above, we will find that some performance criteria conventionally regarded as steady state are in fact set by unsteady flow effects. We will also see the other side of the coin, namely that even in a situation with substantial fluctuations, unsteadiness is not necessarily important by the criterion we are using.

In choosing examples we will hew closely to the rules that we have set up based on the filter described in Section 2. Any illustration of unsteady flow that we show is thus one in which there is an associated overall figure of merit such as efficiency, stalling mass flow rate, peak temperature or other appropriate measure.

4. UNSTEADY FLOW EFFECTS ON COMPRESSOR AND TURBINE AERODYNAMIC PERFORMANCE

Multistage axial flow compressors have been shown to exhibit a higher pressure rise and a higher efficiency when there is close axial spacing between the blade rows compared to the situation when the blade rows are spaced farther apart. Figure 1 (Smith, 1970) illustrates the point. The figure shows data from two tests of a three-stage, low
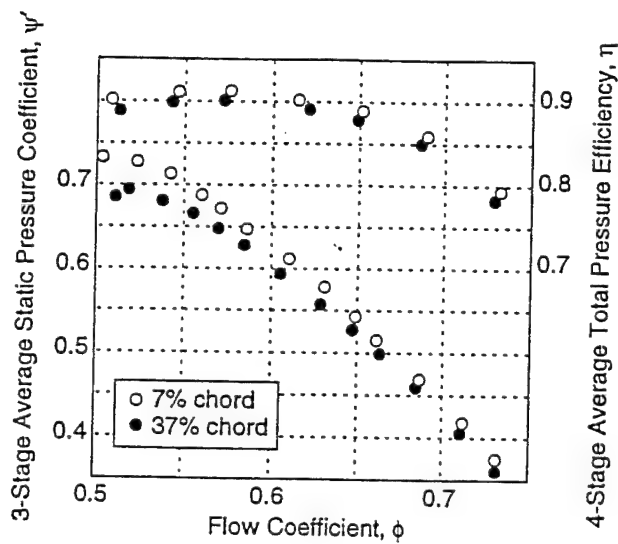
Fig. 1: Multistage compressor performance at different axial spacing (7% and 37% of chord); aspect ratio = 2.10, Re = 178,000, tip clearance = 3% chord (Smith, 1970).

Fig. 2: Passage of stator wake through rotor (after L.H. Smith, 1966).

speed research compressor, one in which the axial spacing was 7% of chord and a second in which the spacing was 37% of chord. There is a one to two percent increase in efficiency and a two to four percent increase in pressure rise for the closely spaced configuration. Data showing a similar trend has also been given by Mikolajczyk (1977).

The data do not show explicitly that the increases are due to unsteady flow, but Smith (1966) has put forward an argument that suggests how such a change could occur. The essence of the argument is given in Fig. 2, which illustrates a stator wake being transported though a rotor. Suppose we consider the wake as inviscid and the fluid as constant density. If so, from Kelvin's theorem the circulation around the contour C remains constant when the wake moves through the rotor, although the length of the contour increases. The velocity difference between wake and freestream thus decreases in inverse proportion to the length. (Comments on wake attenuation from a different perspective have also been given by Ashby (1958).) The loss due to mixing that takes place could thus be less than if the wake had fully mixed before entering the rotor. It would be of interest to carry out a calculation of these two situations using the unsteady codes that now exist to see whether this is the whole explanation or whether one must look elsewhere.

As might be inferred from the vintage of the data, clear demonstrations of these points are few and far between, not only for compressors but also for turbines.
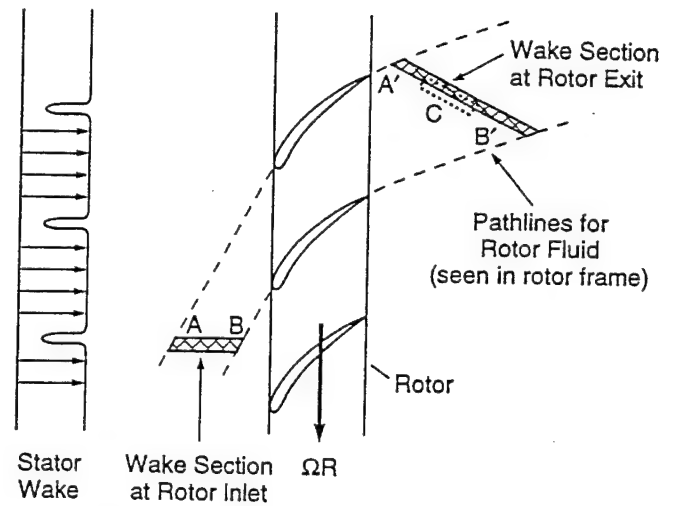
Sharma, Ni, and Tanrikut (1994) show efficiency measurements for a multistage turbine in which the inlet guide vanes were clocked relative to the second stage stators. The efficiency levels, at midspan and overall, versus the clocking position are shown in Fig. 3. It can be seen that there is a roughly one percent difference at midspan and a half percent difference overall depending on the relative
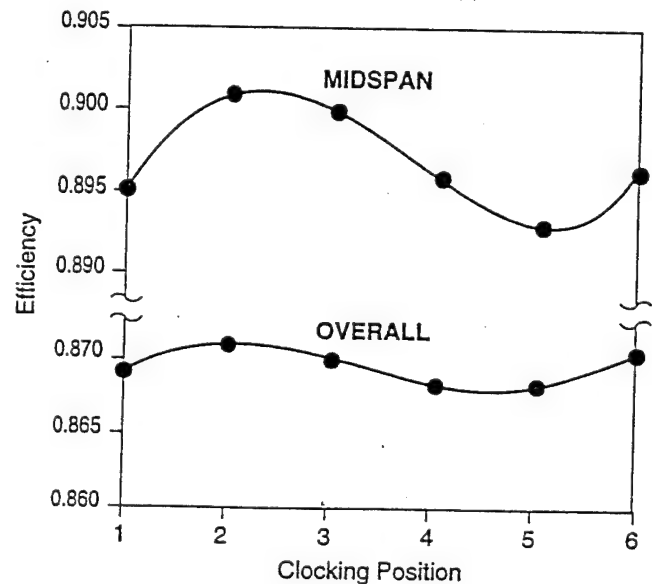

Fig. 3: Turbine efficiency as a function of clocking position (first and second stator) (Sharma, Ni, and Tanrikut, 1994).

clocking location. The physical mechanism responsible for the efficiency variation may or may not be associated with unsteadiness; this has yet to be resolved.

Other examples of unsteady effects on turbine midspan blade row loss and midspan heat transfer have been given by Sharma et al. (1988), Doorly , Oldfield, and Scrivener (1985), and Hodson (1984). These show (see also Mayle, 1991) that wake passage over the suction side of the airfoil is a key cause of transition, leading to both higher losses and higher heat transfer. Figure 4 (Sharma et al., 1988) compares the suction surface heat transfer distribution on a turbine rotor blade operated in a cascade and in a turbine stage environment. The data and calculations show that the boundary layer is transitional and that transition occurs earlier in the stage environment. The relation of this early transition to overall efficiency, however, has not been reported.

5. IMPACT OF UNSTEADY FLOW ON PEAK TURBINE TEMPERATURE LEVELS

A different application in which unsteadiness has been shown to have a significant effect is peak temperature levels encountered in turbines. Figure 5, which is a cut across data given by Sharma, Ni, and Tanrikut (1994), (Sharma, 1994), shows time average rotor surface temperatures in a one-and-a-half stage turbine which is exposed to a hot streak through the inlet guide vane. High temperatures exist on the pressure surface due to the differential migration of hot fluid.
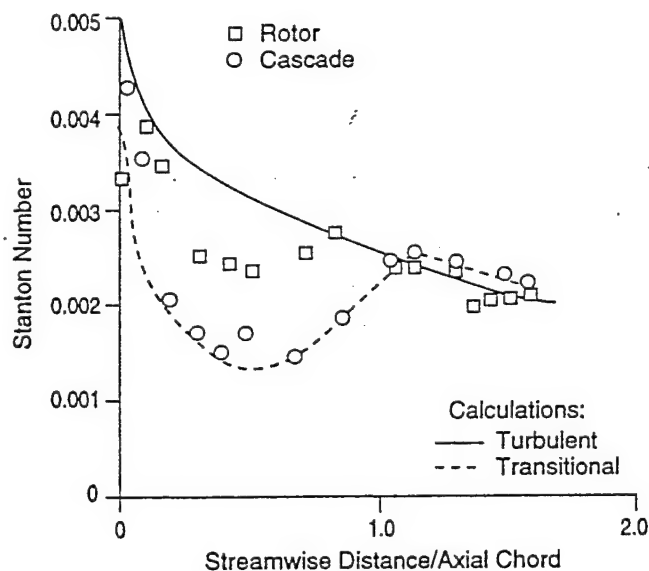

Fig. 4: Stanton number distribution on rotor and cascade (Sharma *et al.*, 1988).
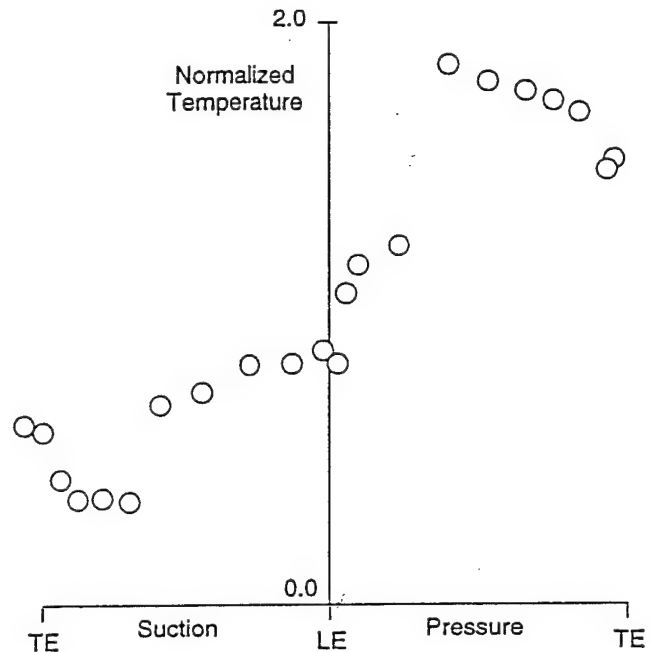

Fig. 5: Temperature difference on the two sides of rotor airfoil due to incoming hot streak (Sharma, Ni, and Tanrikut, 1994; Sharma, 1994).

The basic phenomenon can be qualitatively understood from kinematic (velocity triangle) arguments. At the guide vane exit, the fluid particles in the hot streak have roughly the same Mach number and absolute flow angle as would particles in a flow which had no hot streak. The absolute velocity for the hotter fluid is thus larger (by the square root of the temperature ratio), leading to a migration toward the rotor pressure surface. It may be noted that this type of migration will occur whether the temperature non-uniformity is in the radial or the circumferential direction, but it is only events associated with the latter, i.e. the unsteady processing of the temperature non-uniformity by the rotor, that we discuss here. For this situation, rectification of the periodic impingement of hot fluid on the pressure side gives rise to a higher time average surface temperature.

The kinematic arguments give insight and qualitative information concerning trends with parameters such as design C_x/U. Quantitative assessment, however, can only be achieved by experiment or by true unsteady computations, which can show the rectification that occurs. The calculations that have been done demonstrate that the time average values given by unsteady computations are significantly higher than those from computational procedure in which the incoming flow to the blade row is taken as axisymmetric.

6. UNSTEADY FLOW AND AEROMECHANICAL EXCITATION

There is only sparse information in the open literature concerning the boundaries that mark the onset of flutter (self-excited aeromechanical instability) or levels of forced response vibration (response of a part to an external aero-dynamic force at a resonant frequency of the part). However, some measure of the importance of this topic can be inferred from a recent report on failures in military engines (Air Force Scientific Advisory Board, 1992).

During the past three decades, several hundred incidents ranging in severity from Class A Mishaps (mishaps resulting in a total of one million dollars or more in property damage, or a fatality or permanent and total disability, or destruction or damage of the aircraft beyond economical repair) down to maintenance actions have occurred involving titanium parts in Air Force aircraft engines. High cycle fatigue has been the predominant cause of these, although in a number of cases the receptivity to high cycle fatigue is set up by high steady-state stress. Of the two causes of vibratory stresses, flutter and forced response, it is the latter that appears to be the more important source of high cycle fatigue.

During this time, the increased fidelity of the computational models of structures has led to refinements in design techniques for steady structural loadings. Designers use these refinements to increase mean stress levels and hence reduce weight of the components. However, the increases in mean stress may have reduced (to an undefined extent) the high cycle fatigue life margins available.

In many cases, the critical vibratory frequencies, which are the resonant frequencies of the blades, can be predicted using current structural dynamic computational procedures. On the other hand, the magnitudes of the aerodynamic forcing functions that drive these vibrations have not been well predicted. The magnitude of the structural and aerodynamic damping is neither well understood nor well predicted. On several occasions, unanticipated resonances occurred due to unforeseen forcing functions and/or unexpectedly large variations in structural natural frequencies and damping, which were associated with manufacturing tolerances or parts wear in service. Present design practice is thus to eliminate the most troublesome resonances from the operating envelope as much as possible. The current design trend toward lower aspect ratio blades and higher Mach numbers in compressors tends to exacerbate the unsteady aerodynamic coupling between blade rows and components, and can increase resonant stress caused by aerodynamic interference.

In some instances, there are additional unanticipated phenomena which give rise to aeromechanic problems. In one situation, rotating stall, which was not predicted during design and not seen in the initial testing during development, caused a forced vibration. (The envelopes of the test procedures were altered to reflect the presence of this phenomenon and to ensure that ground tests will screen for rotating stall over the flight envelope.)

Incidents have also been associated with changes in mission definition after completing the design or changing the mission after initial operations, which subjected engines to a harsher environment (with respect to vibratory stresses) than was originally envisioned.

The central question for flutter is the location of the flutter boundaries. There has been impressive progress in the ability to compute unsteady aerodynamics at conditions and geometries that represent the engine environment. So far, however, because of the complexity of the actual flow and the stringent requirements for predictive capability, current systems for predicting flutter are still heavily empirical. Large uncertainties and problems can arise in adapting these empirical criteria to new generations of engines having different geometries.
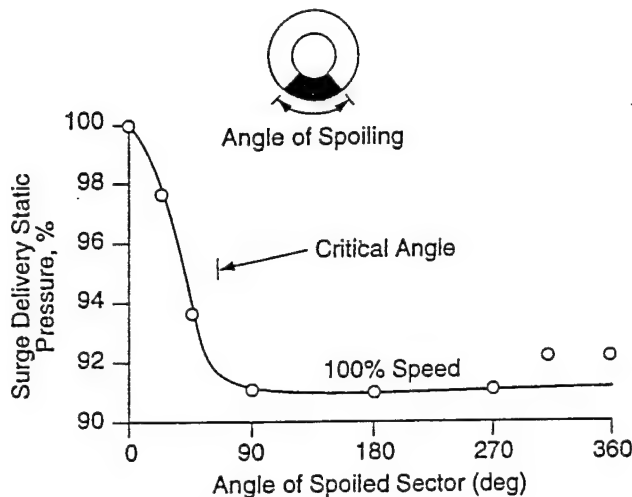
The overall position, therefore, is that there are a number of unsteady aeromechanical phenomena that are of interest to designers. Flutter and forced response are thus issues in which a decrease in the level of empiricism would be of significant value in the engine development process.

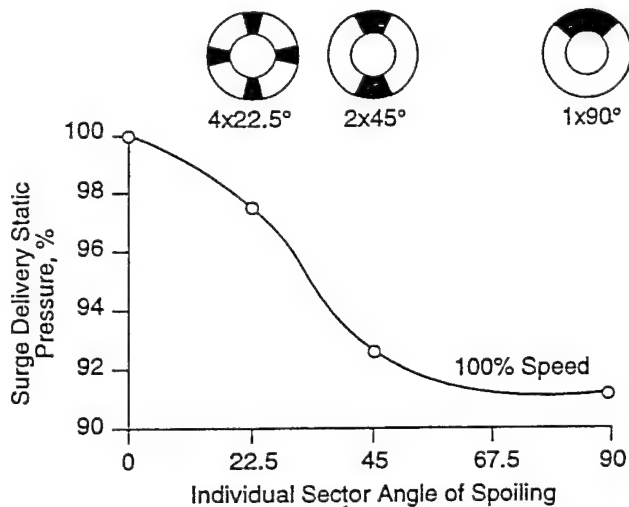## 7. UNSTEADY FLOW AND COMPRESSION SYSTEM INSTABILITY ONSET

The phenomena associated with flow range limitations are inherently unsteady, and in multistage compressors the event that terminates "steady-state" operation is the transition of a small (in some sense) disturbance to large amplitude rotating stall. Details of the various routes to stall and the types of instability have been described at length (e.g. Day, 1993; Tryfonidis et al., 1994) and we do not address them here. What we will do is to show some examples in which the point of instability occurrence has been modified by alteration of unsteady behavior.

The first example concerns compressor response to inlet distortion. Figure 6 shows the response of a six-stage compressor to varying extents of steady-state circumferential inlet distortion (6a) and to a fixed total sector extent which is split into two and four separate sectors (6b) (Bowditch, 1983). The figure of merit is peak pressure rise compared to peak pressure rise with no distortion.

For a steady circumferential distortion, the rotors see the spatially varying nonuniformity as an unsteady flow. In both cases, distortions with higher spatial Fourier harmonics, and hence greater unsteadiness, have the least effect on the stall pressure rise. These results point toward blading designs that take advantage of this unsteady effect,

(a) Effect of Spoiled Sector Width

(b) Effect of Contiguous Spoiled Sector Width

Fig. 6: Effect of: a) circumferential distortion sector angle, and b) number of sectors, on multistage compressor surge pressure ratio (Bowditch, 1983).

i.e. long chord, low aspect ratio, to achieve improved distortion tolerance.

Another facet of the distortion problem is presented in Fig. 7 (Longley et al., 1994). The figure shows theory and experimental results for a multistage compressor subjected to a rotating distortion. (The context for such a situation could be the high pressure compressor of a two-spool engine subjected to distortion from a low pressure compressor which is in rotating stall.) The plot shows the
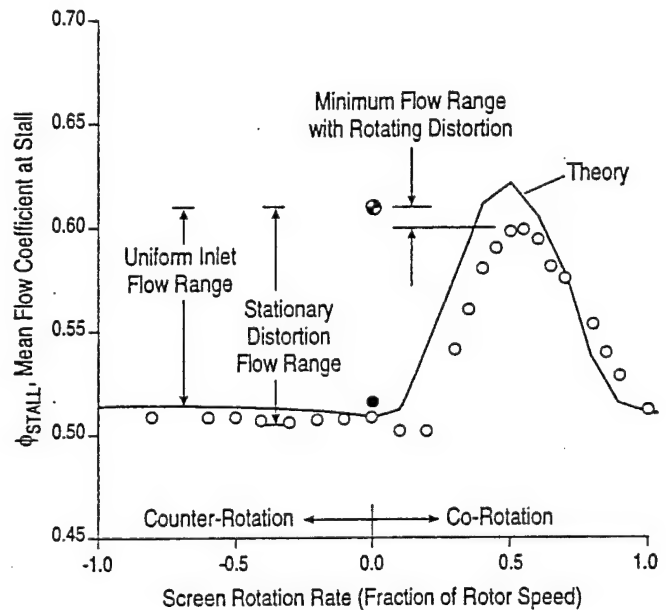

Fig. 7: Decrease in stability with rotating distortion; stall flow coefficient versus distortion rotation rate. ● Uniform flow stall point. ⊗ Design point (Longley et al., 1994).

annulus-averaged mass flow at stall versus non-dimensional distortion rotation speed, and the peak in the response curve near fifty percent speed implies a marked decrease in stable flow range. The theoretical results show that this decrease is associated with forcing the system (in this instance the compressor) at one of its natural eigenmodes. This is a clear instance of a situation in which an unsteady effect causes a large change in a "steady-state" quantity, namely the mass flow at which the machine exhibits rotating stall.

A third example is that of a three-stage compressor with different exit conditions: a constant area annulus, an annular diffuser, and an annular nozzle (Greitzer, Mazzawy, and Fulkerson, 1978). Figure 8 shows that, although the steady-state performance for all three downstream components is essentially identical away from stall, the stall point is changed. It is degraded by the diffuser (compared to the constant area annulus) and augmented by the nozzle. The only way in which the authors can explain this is through the unsteady interaction of the downstream component (basically the downstream impedance) with the compressor.

The same type of behavior is exhibited much more strongly if the downstream component is a compressor, as in the data of Fig. 9 (Williams, 1987), which shows the
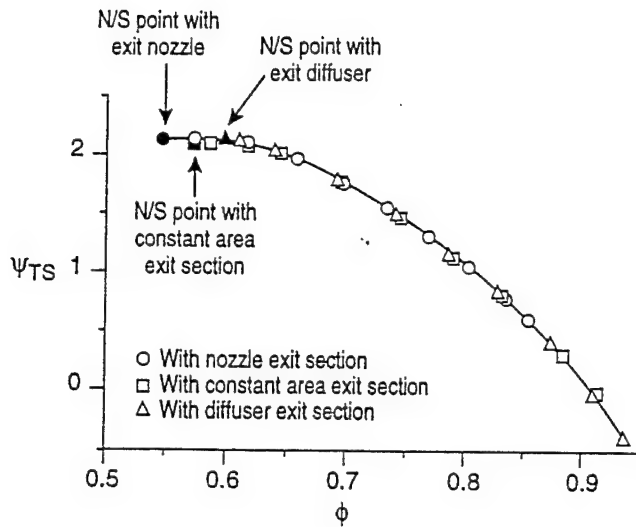
Fig. 8: Effect of downstream components on compressor stability boundary (Greitzer, Mazzawy, and Fulkerson, 1978).

distortion tolerance of a two-spool engine, along with calculations of the stalling component based on assuming different amounts of coupling between components. From analyzing the low pressure compressor in isolation one would predict it to be the component that leads to system instability. Analyzing the two components together, however, shows that the high pressure compressor is responsi-
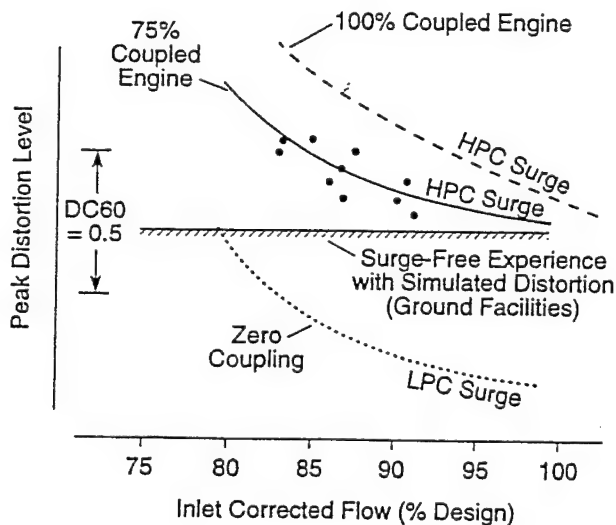
ble. The message to the designer is that the unsteady phenomena associated with stall onset can be altered by the downstream components in ways that are beneficial or detrimental, and that one should design each component as part of a closely coupled system.

The final example in this section is stabilization of rotating stall using active feedback control. The idea is to sense the small unsteady disturbances that can signal the onset of rotating stall, process the signal, and use the processed signal to drive a suitable actuator or set of actuators, as shown schematically in Fig. 10. The combination of compressor plus controller constitutes a new machine with different unsteady flow properties, even though the steady-state behavior may be essentially the same as that for the compressor alone. The basic concepts, flow and controller models, and control strategies have been described elsewhere (Paduano et al., 1993) and we only view the results via a steady-state metric, the overall mass flow at which the compressor encounters rotating stall. This is presented in Fig. 11, which shows the onset of instability with no control and with control of the first three natural modes of the compressor. Knowledge of the unsteady behavior is a key ingredient not only in formulating the concept of active control, but also in implementation of the successful system implied by the results of Fig. 11.

8. OVERALL CLASSIFICATION OF THE UNSTEADY PHENOMENA THAT HAVE BEEN DISCUSSED

We have looked at a number of examples in which unsteady effects can be said to have an impact on some aspect of performance that is of interest to designers. We

Fig. 9: Distortion tolerance for different amounts of coupling between compressors. • Engine data (Williams, 1987).
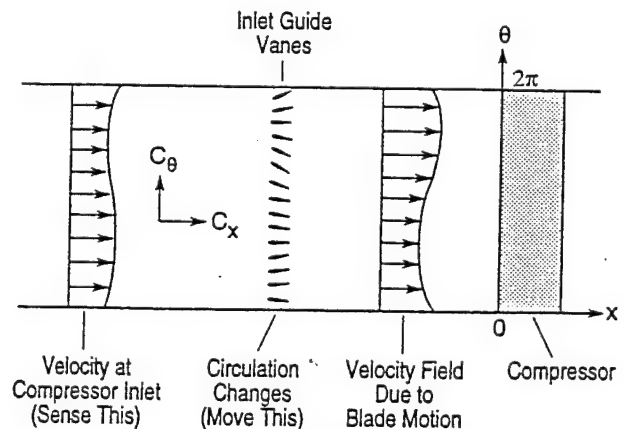

Fig. 10: Conceptual control scheme using "wiggly" inlet guide vanes to generate circumferential travelling waves (axial distances not to scale).
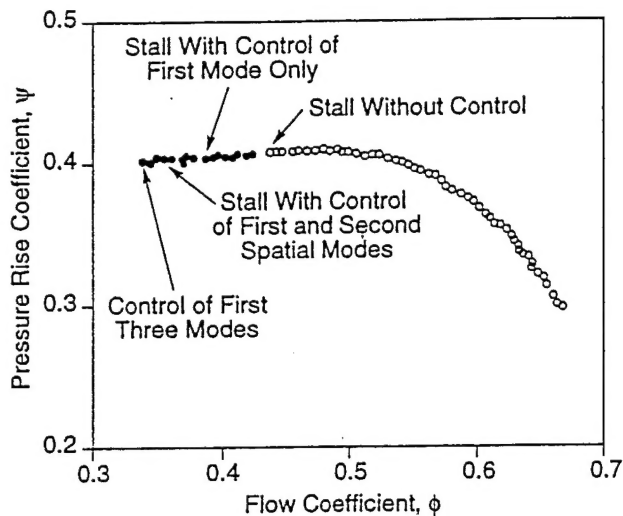
Fig. 11: Increase in stable flow range with active control of single-stage axial compressor (Paduano *et al.*, 1993).

now wish to take a further step and to look towards the identification of some fruitful research topics from the diverse array of unsteady flow events that exist in turbomachines. In doing this, it is useful to classify the phenomena that we have been discussing by length scales, dividing these into three regimes. "Small scale" in this view is unsteadiness associated with wake and boundary layer turbulence and transition, perhaps characterized by the integral length scale which is an order of magnitude or more smaller than the blade spacing or chord. "Medium scale" is associated with the passage flow structure, in other words with phenomena on the order of the pitchwise blade spacing. Examples are upstream potential effects, overall wake velocity distributions, and tip clearance vortices. "Large scale" phenomena have length scales roughly an order of magnitude larger than the blade passage. Flutter, low engine order forced vibration, rotating stall, and surge are examples of this type.

The classification is useful to bring out an overall conceptual point. For the small and large scale unsteady phenomena, the relative importance (or lack of importance) on overall performance has been established in a number of situations and there are well-documented causal links between unsteadiness and the figures of merit mentioned previously. In many instances, there exist estimates of what amount of benefit might be associated with management of the unsteady effects. This is true even though these phenomena are often complex enough so that only the qualitative features are known and we cannot calculate

them with adequate accuracy.

This is, however, not the case with the medium scale, passage scale, phenomena. Here the causal relationships are not clear, nor is there much quantitative information on the level of benefits to be expected at present. We suggest below some procedures to address this lack of knowledge.

9. SPECIFIC RESEARCH ISSUES IN UNSTEADY FLOW

With the above classification in mind, we now address some specific areas of research. In doing this, we draw on the discussions at the NASA symposium (Adamczyk et al., 1992) and the AFOSR sponsored workshop (Fant and Murthy, 1993) on unsteady flows in turbomachines.

For small length scale phenomena the issues are turbulence and transition. One problem that was singled out at the NASA meeting was effects of flow transition on turbine performance, although one might make a reasonable case that semi-empirical methods now exist to estimate the location of transition with sufficient accuracy for design (Mayle, 1991). The consensus of the NASA meeting was also that much of the transition work currently being carried out by the fluid dynamics community is aimed at aspects of the external flow problem and the route to turbulence in a gas turbine engine, which is strongly dependent on the upstream forcing, is quite different. Knowledge of the turbulence structure, by itself, does not seem to be a limiting item for design, at least for the near future.

The impact of large scale unsteady phenomena has been stated several times. In some instances the same situation (e. g. inlet distortion, rotating stall) can give rise to both aeromechanical and aerothermal difficulties. Thus, effects of distortion have important consequences as far as forced response, aerodynamic performance, and stall margin are concerned. In this connection, there currently exists no rigorous method for predicting the onset of flutter or of aerodynamic instability. Development of either of these would be a significant advancement. Development of reliable methods for assessment of resonant stresses is also an issue of import. Some aspects of this problem are yielding to computations (for example upstream effects of the potential field of the downstream blade rows) but there are many other less well-defined sources of forcing which need illumination. Also included in this category is the new paradigm of active control of unsteady flows. This offers not only possibilities for mitigating the consequences of aerodynamic and aeromechanical instability, but also totally new diagnostic procedures for exploring the dynamic behavior of compresson systems.

The selection of topics and issues is least clear for the medium scale phenomena. Again the question of aeromechanical response is known to be important. The consequences of rotor-stator interaction (forced response due to

both wakes and to upstream potential effects) are not well enough defined such that one can design with confidence, but computational methods are now appearing (e.g. Manwaring and Wisler, 1993; Rao, Delaney, Dunn, 1994; Giles, 1991) that should allow one to attack this problem in a rigorous manner.

There are a number of other areas, however, which require further definition of the problem before its importance can be assessed. For multistage turbomachines, for example, an important problem is how the information is passed between stationary and rotating rows. It is not clear at present how important the non-axisymmetric (and hence unsteady) effects are compared to those which are axisymmetric in nature. To find this out requires implementation of, in Marble's (1991) words, "simple, clear, examples" which carry through from the basic phenomena that are being investigated to the overall performance. This is now becoming feasible, and is an item of research that we strongly support.

A general comment is that unambiguous demonstrations of the role of unsteadiness are needed to point out specifically where medium length scale unsteadiness is important. This has been done in a few cases, perhaps most successfully the turbine wake transport and hot streak migration problem (e.g. Krouthen and Giles, 1988; Dorney, Davis, and Edwards, 1990) This is essentially a model problem in that there is an agreed upon research agenda, well-differentiated levels of detail at which the problem should be attacked, and an identifiable benefit. There are other areas, however, in which one could resolve this question by executing some well thought out numerical experiments, and we will mention some of these.

The effects of rotor-stator interaction on efficiency is a topic that has been discussed for many years, but even now there is little direct evidence on the point. Present computational procedures have the ability to address this and it is suggested that efforts be made to obtain more definitive statements about the role of unsteadiness, both two- and three-dimensional. The interactions between impeller and diffuser in a centrifugal compressor is an area that can definitely benefit from some well thought out numerical experiments to assess quantitatively the impact of unsteadiness.

A final topic is the manner in which the tip and hub flows (including cavities), both of which are strongly three-dimensional and non-uniform in the circumferential direction, are processed by the downstream row. Does it matter that the flow is unsteady as far as peak pressure rise and efficiency are concerned? The data shown earlier implies that it does, but what is the mechanism and can anything be done about it?

10. COMMENTS ON A VISION FOR FUTURE WORK ON UNSTEADY FLOW IN TURBOMACHINES

Many of the questions posed in the last section could be resolved with computational procedures that either exist now or are being created. From the designer's point of view, the focus of the computations should be on answering some well-defined fluid dynamic question. In this connection, we stress that there are many questions that are generic, in that they apply across a reasonably wide range of design space. Obtaining answers as to whether an effect is important (by the standard we have adopted) should not be critically dependent on having geometry that is an exact replica of any specific proprietary configuration.

It is our belief that a key aspect of future work in this field is conducting careful, well thought out, numerical experiments to furnish the estimates needed to answer which of the various middle length scale phenomena are important to the design process. This is an opportunity which has not been available up to now. We can illustrate perspective on this use of the capabilities that exist now by an example concerning unsteady flow round a compressor blade. The circulation around the airfoil is not steady but fluctuates periodically due to the neighboring blade rows, with the result that a vortex sheet of fluctuating strength is shed. This has long been recognized, and mixing out of the non-uniformity in velocity which characterizes the vortex sheet has been mentioned as one possible source of loss. Calculations to examine this issue have recently been carried out by Fritsch and Giles (1992) and their conclusion is that this mechanism will be responsible for roughly 0.03% loss in efficiency in a representative transonic compressor geometry, in other words that it has a negligible influence on design choices. This does not mean that the result is not interesting nor that the calculation was not worth doing. Far from it, it is one step in the attempt to focus on those pieces of this complex puzzle that should be included in a description of the unsteady flow field. Other examples in which unsteady computations have been used to sort out the time-averaged effects on loss and efficiency are the work of Valkov (1992) and of Adamczyk, Celestina, and Chen (1994). We view work of this type as important, useful, and instructive.

We close with brief remarks on the process of defining research topics and objectives. If the goal is to transition the results of unsteady flow research into the aeroengine design process, we suggest that considerable forethought be given to the context in which the specific fluid dynamic problem is posed. This includes not only the use to which the results will be put but also the coupling with other disciplines which may constrain the design. Some questions

relating to this process and to advocacy for unsteady flow research are given in the Appendix. Further, for the transition process to be most effective, there must be significant and on-going technical contact between academia and government laboratories and industry, who are the eventual customers for the technology. It is our experience that while management in these organizations can foster a climate in which such contacts are encouraged, there must also be a willingness of individuals on the working level to engage in substantive technical dialogue, rather than to stay isolated in their separate camps. Although this represents, in a large number of situations, a new way of doing business as far as research is concerned, there is no substitute for such communication. It is often difficult to open up these pathways, but it has been the experience of the authors that close connection with "real world" aeropropulsion problems is an extremely effective means to generate research that is not only technologically relevant but scientifically fruitful and intellectually challenging .

11. SUMMARY AND CONCLUSIONS

A number of unsteady flow phenomena are inherent to turbomachinery operation. The consequences on machine operation range from negligible to performance-limiting phenomena. Although understanding of many unsteady processes is still in a rudimentary state, this has not prevented the development of efficient and robust turbomachines. In this paper, we have attempted to show that there are many areas in which unsteady flow directly impacts turbomachine performance as well as operability and reliability. In these areas, one can find tangible benefits from increased understanding of unsteady flow and integration of this understanding into the design process. We have also shown that, from the point of view of a designer, there are other areas in which a strong case for the impact of unsteady flows has not been made, although the tools are now at hand to resolve the issue. Suggestions are given for research to be carried out on both of these fronts with the objective of enabling the aeroengine designer to predict and to manage those aspects of flow unsteadiness that do contribute significantly to the achieving of design goals.

12. ACKNOWLEDGMENTS

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APPENDIX
COMMENTS ON THE PROCESS OF ADVOCACY FOR UNSTEADY FLOW RESEARCH

It is our view that to forge a truly solid case for research in unsteady flow it is useful to address some basic questions concerning the topics proposed for study. One list of these might be:

- What should we study and why?
- How can projects be selected to maximize potential for integration into a design approach?
- How "detailed" an understanding are we seeking or do we need?
- How "fully" is "need to fully account for"?
- What does "major advancement" mean?
- Do the participants understand design?
- How will university/industry/government interact to effectively transition the results to products?

Carrying out the exercise of providing responses to such questions has strong technical as well as advocacy benefits. In particular, it allows:

1) Identification of research topics that designers think have high eventual payoff. This can spawn academic interest, if it does not already exist, which can lead to unsteady flow research.

2) Identification of topics which are viewed by designers as having little practical payoff.

3) Identification of topics for which there may be little or no present design interest, but high academic interest as well as interest from an enabling technology context.